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TECHNIQUES FOR THE DETERMINATION OF MASS PROPERTIES OF EARTH-TO-ORBIT TRANSPORTATION SYSTEMS

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THE UNITS FOR THE PHYSICAL QUANTITIES DEFINED IN THIS PAPER ARE GIVEN BOTH IN THE INTERNATIONAL SYSTEMS OF UNITS (SI) AND IN THE U.S. CUSTOMARY UNITS. TABLES II AND III ARE IN U.S. CUSTOMARY UNITS. COMPUTER RESULTS HAVE BEEN PROGRAMMED FOR THE PRINTOUT IN BOTH UNITS. THE MEASUREMENTS AND CALCULATIONS WERE MADE IN U.S. CUSTOMARY UNITS.

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SUMMARY

As a possible follow-on Earth-to-orbit transportation system, single- and two-stage winged vehicles are being studied. All propellants are carried internally and the vehicle returns to base for an aircraft-type landing. Such studies require a reasonably accurate means of rapidly determining the mass properties of the overall system when various vehicle design parameters are varied. Two techniques have been developed; one involves the trending of the overall vehicle to a new size when mass properties are already known from a prior detailed analysis; the other technique involves trending each subsystem from known space shuttle, aircraft, and applied research hardware to determine overall mass properties by summation of the trended subsystems of vehicles for which little is known initially.

Several fairly extensive documents for mass estimating have been published for two-stage fully reusable and single-stage-to-orbit systems (refs. 1 through 6). The intent of the present work is to extend present capabilities to emerging new classes of vehicles. This has been achieved by modifying these equations which were originally intended for commercial or fighter aircraft for the Earth-to-orbit vehicles--vehicles which are markedly different in structural concept and material usage.

SYMBOLS

I. Vehicle Geometry

L_r	=	vehicle reference length, m(FT)
L_w	=	exposed structural wing span, m(FT)
L_b	=	body width at wing-body juncture, m(FT)
S	=	vehicle total planform area, m^2 (FT^2)
S_b	=	body planform, m^2 (FT^2)
S_f	=	body flap planform, m^2 (FT^2)
S_c	=	total control surface planform (includes body flaps, elevons, and rudders) m^2 (FT^2)
S_t	=	tail profile area, m^2 (FT^2) including rudder
S_{tk}	=	tank area, m^2 (FT^2)
S_w	=	exposed wing planform, m^2 (FT^2)
S_{wet_v}	=	total vehicle wetted area, m^2 (FT^2)
S_{wet_b}	=	body wetted area, m^2 (FT^2)
T_r	=	exposed wing root chord, max. thickness, m (FT)
$\bar{t}$	=	equivalent thickness of support structure for RSI, or thermal capacity factor, mm (IN)
t	=	tank wall thickness, mm (IN)
V_p	=	pressurized volume including crew and wheel compartments, m^3 (FT^3)
K_{v1}	=	vehicle trending point design technology/configuration constant, Gg (MLb)
F	=	the dimensional ratio of the off-point design wing to the point design, dimensionless

II. Subsystem Masses (all units in this section are in Kg or Lbm)

m = total propellant mass in vehicle

m_a = avionics mass

m_{ARES} = ascent residual and reserve mass

m_b = body mass including tanks

m_d = vehicle dry mass less margin

m_{DES} = mass of vehicle at descent (i.e., mass of vehicle after execution of deorbit maneuver)

m_e = vehicle mass at "entry" i.e., at 122 Km (400,000 ft) altitude after depletion of entry attitude control propellants

m_{ENG} = mass per main engine

m_f = estimated fixed masses (total) in the vehicle including payload, cargo bay doors and structure, manipulator, avionics, crew compartment and crew

m_c = on-orbit and deorbit attitude control and maneuver propellant

m_g = landing gear, manipulator, and docking system mass

m_{gr} = gross mass

m_{INJ} = vehicle mass at injection

m_I = inert mass of vehicle, or gross less fixed masses and ascent propellant (subscripts 1 and 2, point design and off-point design vehicles)

m_L = landed mass

m_{ma} = manipulator mass

m_s = docking and separation system mass

m_{pt} = total propellant mass

m_p = ascent propellant mass

m_{p_1} = ascent propellant mass in point design vehicle

m_{p_2} = ascent propellant mass in off-point design vehicle

m_w = wing mass

m_{sc} = surface control mass

m_{env} = environmental control system mass

m_{mar} = growth mass

m_{TPS} = thermal protection system mass

m_{w_u} = unit wing structural mass

m_t = vertical tail mass

m_o = OMS maneuver system mass

m_r = all up reaction control system dry mass, including engines and tanks

m_{el} = electrical subsystem mass

m_s = separation and docking system mass

m_{pf} = maneuver engine pressurization and feed

m_R = reusable surface insulation average unit mass over the entire vehicle

m_{OP} = OMS propellant mass

m_1&m_2 = unit masses of point design and off-point design wing respectively based on exposed planform

m_{av} = avionics mass

m_{uf} = residual and unusable fluids

m_m = maneuver engine mass

m_{pow} = prime power mass

m_{ROMS} = mass of OMS reserves

m_{RRCS} = mass RCS reserves

m_{orr} = OMS + RCS reserves

m_{RCS_e} = entry RCS propellant

m_{INF} = inflight losses

m_h = hydraulics system mass

II. (CONT'D)

GLOW = gross liftoff mass

III. Mission

ΔV_{IDEAL} = delta "Vee" equivalent for total mission, m/sec (ft/sec)

ΔV_{ro} = delta "Vee" equivalent for attitude control on-orbit, m/sec (ft/sec)

ΔV_{o} = delta "Vee" equivalent for maneuver system, m/sec (ft/sec)

ΔV_{re} = delta "Vee" equivalent for attitude control entry, m/sec (ft/sec)

IV. Propulsion Performance

$\dot{m}$ = mass flow per engine, Kg/sec (lb/sec)

T_{VAC} = vacuum thrust per main engine, Newtons (lbf)

$I_{s_{re}}$ = average reaction control system specific impulse during reentry, sec

$I_{s_{ro}}$ = specific impulse of reaction control engines on-orbit, sec

I_{s_m} = specific impulse of the OMS maneuver engine degraded for the estimated number of restarts, sec

I_{s_e} = main engine vacuum specific impulse, sec

λ_1 = performance mass fraction or ascent propellant divided by gross mass for the point design vehicle (dimensionless)

λ_2 = the performance mass fraction of the off-point design (dimensionless)

λ_1' = point design trending propellant mass fraction (or ascent propellant divided by gross mass less fixed mass) dimensionless

M.R. = mass ratio equals gross lift-off mass divided by burnout mass (dimensionless)

V. Subsystem Constants: Paragraph numbers below correspond to items in mass properties tabulations, Tables II and III, and the suggested format of the Mil Spec (ref. 12)

1.0/2.0 Wing and Tail

W_m = Wing material/configuration constant

W_c = Wing carry-through material/configuration constant

V_t = Material/configuration tail constant

3.0 Body

B_c = Crew cabin constant

B_b = Body structure constant

B_{bf} = Body flap

B_f = Fuel tank constant including insulation and non opts

B_o = Oxidizer tank constant including insulation and tank non opts

4.0 Thermal Protection System

K_r = A constant for average TPS mass, based on wetted area

5.0 Landing Gear

K_L = Function of landed mass

6.0 Main Rocket Engine

R_{ph} = Engine power head

R_n = Nozzle

R_{ne} = Nozzle extension

R_{na} = Nozzle extension actuator

R_{pf} = Pressurization and feed system

Subsystems Constants (Cont'd)

R_{ga} = Engine girbal actuator

ε = Expansion ratio

7.0 Reaction Control System Constants

R_{RCS} = Overall system constant

8.0 Maneuver Engine Constants

M_{me} = A point-design constant for the maneuver engine (includes nozzles, actuators, etc.)

M_t = Tank system constant for maneuver propellants

M_{pf} = Pressurization and feed system constant for maneuver system

9.0 Auxiliary Power Constants

A_{ap} = Auxiliary power unit constant

A_{ac} = Actuator system constant

Power Subsystem Constant

PW_b = Battery power demand constant

PW_e = Engine power demand constant

PW_c = Surface control power demand constant

10.0 Electrical Conversion and Distribution

E = A constant for generators and wiring (does not include generator drive)

11.0 Hydraulic Conversion

H_{cs} = Control surface power constant

H_e = Engine gimbal system power constant

12.0 Control Surface Constants

S_{sc} = Aero surface control constant

S_{pc} = Pilot related controls

13.0 Avionics

M_{av} = avionics mass constant

14.0 Environmental Control

E_c = Pressurized volume constant

E_o = Oxygen supply constant

E_a = Avionics heat load constant

15.0 Personnel Provisions

PP_f = Food, waste, and water management systems

PP_s = Seats and other pilot and crew related items

16.0 Margin

MAR = a percentage of dry mass less engine mass to allow for growth uncertainty

17.0 Personnel (i.e., crew and mission specialists)

P_p = mass of individual personnel including personal gear, life support, and crew accessories

20.0/27.0 Fluids

R_{rf} = Residual fluids

R_o = OMS reserves constant

$R_{p\ell}$ = pre-launch losses and engine thrust build-up

R_r = RCS reserves constant

R_{ar} = ascent reserves constant

R_{ap} = ascent propellant residuals constant

R_{inf} = inflight losses constant

VI. Miscellaneous

K_{tK} = tank area constant

r = tank radius (m, FT)

V_2 = volume of LH_2 tanks, or fuel tank (m^3, FT^3)

V_1 = volume of LOX tanks, or oxidizer tank (m^3, FT^3)

λ_p = packaging efficiency or body propellant volume/total body volume (dimensionless)

F = ratio of dimensions of photographically enlarged vehicle to point design vehicle (dimensionless)

g = gravity constant, m/sec^2 (FT/sec^2)

C_L = average lift coefficient during entry, dimensionless

D = days in orbit

e = 2.718 (constant)

e_g = radius of gyration proportionality constant

f = body wing efficiency factor and is the ratio of lift intensity on body to lift intensity on wing (f = 0.2 for conventional vehicle to 0.15 for control configured vehicle)

P_c = main engine chamber pressure, N/cm^2 (Lb/in^2)

P_u = tank ullage pressure, N/m^2 (Lb/in^2)

N_z = load factor equals safety factor X ultimate load factor.

N_c = number of crew, mission specialists, and passengers

N_e = number of engines

ρ = tank wall density (Kg/m^3, Lb/in^3)

σ = tank wall limit stress (N/cm^2, Lb/in^2)

R_{PL} = pre-launch losses

I. INTRODUCTION

Two techniques for determining the mass properties of space transportation systems are presented in this report; namely, overall vehicle trending which requires little detailed information about a given vehicle and subsystem trending which requires more detailed subsystem analyses. A single-stage-to-orbit rocket powered vehicle (designated EH 155) is used extensively as an example in discussing both techniques. The vehicle is an in-house design. It is launched vertically but is provided with wings for a horizontal landing (fig. 1).

For each of the two techniques, sample computer results have been included (Appendices A and B). For the overall vehicle trending case, a sample problem and the computer tabulated results have been included (Appendix A). The method is useful for rapidly obtaining projections of vehicle performance as a rocket stage when vehicle size is altered. The savings in computer input time is substantial since only about six inputs are required for this method which is based on knowledge of the way in which subsystem masses vary as a function of vehicle size (table I).

When a more detailed analysis is required, each subsystem is trended and the summation of all the subsystem masses gives the overall vehicle mass. In this regard, the shuttle orbiter subsystem characteristics have been utilized extensively to establish constants for "current technology" subsystems. For this reason, a sample test case of this vehicle has been included which shows the actual current masses of the various orbiter subsystems configured with the Systems Engineering

Mass Properties Program (SEMP) (table II). For various reasons, it is not intended that the program results check "exactly" with the shuttle orbiter since these latter masses vary with subsystem maturity, and in some case added technology leverage, applied as the shuttle program progresses.

For both the shuttle orbiter and single-stage-to-orbit vehicle (EM-155), a test case is shown in Appendix B while sample computer inputs for this program have been included in Appendix C. Utilizing the shuttle subsystems as a starting point, other program input constants have been established (table III). The constants (other than shuttle) usually represent utilization of some type of advanced material, method of construction, or other developmental subsystem.

II. OVERALL VEHICLE TRENDING - DETAILED DESCRIPTION OF METHODOLOGY

When a point design vehicle mass is established, a trending technique is useful for determining masses for other vehicles without analyzing the new vehicle subsystem-by-subsystem. Only six computer inputs are required to determine overall mass properties of the vehicle. This technique is useful when two-stage systems are involved and the relative sizes of the orbiter and booster are being varied in order to optimize the system. Likewise the technique is useful when resizing single-stage systems to meet varying mission and design requirements.

The trending technique is based on the knowledge that for moderate changes in vehicle size most of the subsystem masses vary in accordance with some exponential related to vehicle dimension. For instance, the mass of the thermal protection system is a linear function of vehicle wetted area assuming the entry profile and entry planform loading are not too different. In terms of vehicle length, this subsystem varies as L_r^2 (where L_r is vehicle reference length).

Main propulsion system tank mass, if nonintegral, is much more sensitive to vehicle physical size and varies as L_r^3 or directly as propellant mass since tank mass is approximately equal to a constant times propellant volume for any given shape. (This can be proven from the basic relationships for tank volume, wetted areas, and hoop stresses.) The assumption is inaccurate to the extent that insulation weight is proportional to tank wetted area or dimension squared (not cubed) and tank nonoptimums decrease slightly as size increases. Further, if the tank is integral and carrying body loads, most of the tank wall will be designed by compressive limit crippling loads and not pressure.

When the vehicle's rocket engine system is similarly assessed, these masses vary as L_r^3. For example, doubling the vehicle's length yields eight times the propellant tank capacity or very nearly eight times the lift-off mass. If the same lift-off thrust-to-mass ratio is to be maintained to maintain similar performance, then eight times the engine thrust (hence mass) is required; engine mass being approximated by a constant times thrust. This assumption is inaccurate to the extent that dry mass is a slightly decreasing percentage of gross mass.

Similar logic is applied to other subsystems (table I) identifying the exponential most applicable to the given subsystem. Assuming all the dry masses vary as L_r^2 (reference length squared) and propellant mass as L_r^3 (volume) then the relationship between vehicle mass and propellant loading for point designs 1 and 2 can be shown to be:

For the propellant masses:

$$\frac{m_{p_2}}{m_{p_1}} = \left[\frac{L_{r_2}}{L_{r_1}}\right]^3 \tag{1}$$

and for the inert (or "dry" mass):

$$\frac{m_{I_2}}{m_{I_1}} = \left[\frac{L_{r_2}}{L_{r_1}}\right]^2 \tag{2}$$

or

$$m_{I_2} = \left[\frac{m_{p_2}}{m_{p_1}}\right]^{2/3} m_{I_1} \tag{3}$$

For simplicity of analysis inert and dry mass are used here interchangeably; M_I includes all subsystem dry masses and a small percentage of residual fluids, both of which are proportional to vehicle size.

As stated earlier, external thermal protection system mass and other major subsystems vary as L_r^2; however, the main rocket engines and tanks vary as L_r^3. If all subsystems varied as L^2 an exponential of two-thirds in equation (3) would be valid; but if all subsystems varied as L^3 then the exponential of unity would be exact. It has been found from actual tests of the equations that an intermediate value of five-sixths gives general agreement with actual detailed vehicle designs (see subsequent discussions in this section for sensitivities and vehicle comparisons). Equation (3) then becomes:

$$m_{I_2} = \left[\frac{m_{P_2}}{m_{P_1}}\right]^{5/6} m_{I_1} \qquad (4)$$

Another aspect of trending involves provisions in the resulting equation for subsystem masses which do not change appreciably with vehicle size. For instance, for a given mission, crew complement would remain constant; typically, avionics and environmental control remain fairly constant. For a constant volume cargo bay, doors and structure remain fairly constant. This leads to the necessity for provisions in the trending equation for the elements hereinafter referred to as "fixed" mass.

When separating the fixed mass the equation for the performance mass fraction can be written as propellant loading divided by gross mass which consists of main propellant, fixed masses, and inerts, or:

$$\lambda_2 = \frac{m_{P_2}}{m_{P_2} + m_f + m_{I_2}} \tag{5}$$

Now the concept of trending mass fraction is introduced, which equals the main ascent propellant loading divided by the inerts plus propellant. In this definition fixed masses are again excluded, or:

$$\lambda_1' = \frac{m_{P_1}}{m_{I_1} + m_{P_1}} \tag{6}$$

or solving for m_{I_1}

$$m_{I_1} = \left[\frac{1 - \lambda_1'}{\lambda_1'}\right] m_{P_1} \tag{7}$$

substituting the value for m_{I_1} in equation (4) it becomes:

$$m_{I_2} = \left[\frac{m_{P_2}}{m_{P_1}}\right]^{5/6} \times \left[\frac{1 - \lambda_1'}{\lambda_1'}\right] m_{P_1} \tag{8}$$

substituting this value in turn for m_{I_2} in equation (5) and dividing numerator and denominator by m_{p_2} it becomes:

$$\lambda_2 = \frac{1}{1 + \frac{m_f}{m_{p_2}} + \left[\frac{m_{p_1}}{m_{p_2}}\right]^{1/6} X \frac{1 - \lambda_1'}{\lambda_1'}} \qquad (9)$$

In this equation, the propellant loading, m_{p_1}, and the trending propellant mass fraction, λ_1', are constant for a given point design and may be treated accordingly,

$$\text{or:} \qquad K_{V_1} = \left(m_{p_1}\right)^{1/6} X \frac{1 - \lambda_1'}{\lambda_1'}$$

which is henceforth defined as the point design technology/configuration constant.

Equation (9) for the off-point design then becomes:

$$\lambda_2 = \frac{1}{1 + \frac{m_f}{m_{p_2}} + \left[\frac{1}{m_{p_2}}\right]^{1/6} K_{V_1}} \qquad (10)$$

In equation (10) above, as propellant mass, m_{p_2}, of the new point design increases so does the performance mass fraction, λ_2. In effect, the "fixed" masses are becoming a smaller and smaller percentage of propellant loading, mp_2. "Growing" the vehicle then is one means of improving design performance. The one-sixth exponential in equation (10) is the result of the assumption of five-sixths in equation (4). The sensitivity to two-thirds power rule and cubed rule can be seen in upper and lower curves respectively in figure 2 wherein the final equation yields exponentials of one-third and zero. The difference in estimated propellant loading between one-third and one-sixth exponential is approximately 1 percent for the due-east mission (see dotted horizontal line in figure 2).

To obtain the values for mass ratio in figure 2 the definitions for performance mass fraction and gross mass for the off-point design are applied, namely:

$$M_{gr} = \frac{Mp_2}{\lambda_2} \tag{11}$$

and

$$M.R._2 = \frac{M_{gr}}{M_{gr} - \lambda_2 M_{gr}} = \frac{1}{1 - \lambda_2} \tag{12}$$

For a booster, fixed masses are small since no payload is carried within the vehicle, and due to its physical size, crew and avionics would normally be small compared to vehicle mass. With little error, when applied to a booster, equation (9) can be reduced to:

$$\lambda_2 = \frac{1}{1 + \left[\frac{m_{P_1}}{m_{P_2}}\right]^{1/6} \times \frac{1 - \lambda_1}{\lambda_1}} = \frac{1}{1 + \left(\frac{1}{M_{P_2}}\right)^{1/6} K_{V_1}} \qquad (13)$$

In the equation above, λ_1'has become λ_1, performance mass fraction, since no fixed masses are assumed.

Reducing the point design technology/configuration constant, K_{V_1}, in equation (10) or (13) is another alternative to improving performance as opposed to "growing" the vehicle which was mentioned previously. This constant is strongly influenced by the extent to which emerging technologies are applied to body and wing structure. It is also influenced by configuration, particularly cargo bay shape and size, and the ingenuity of the designer in packaging the vehicle for the least amount of required structure.

In the design process, a required propellant loading is estimated. When the structural analysis has been completed, or the subsystem elements weighed by simpler means, such as subsystem trending, it is found that the vehicle either has inadequate or excessive performance. Equation (10) is then useful to simply trend the vehic to the required propellant loading from the point design with much reduced input time.

Two plots are shown for equation (10) in figure 3. The point design A is shown having a performance mass fraction λ_1 of .876 or a mass ratio (M.R.) of 8.08 for a payload requirement of 29,500 Kg. If detailed analysis should show that the inert mass of the vehicle has increased, then a new point design "B" of the same propellant loading results. This new generic design trends along a line of lower mass ratios for any given propellant loading. In order to restore the vehicle to the same performance line, the vehicle must be trended (i.e., "GROWN") to point C. Any other changes in basic configuration, materials technology, or other would be considered as a change in point design and would trend along a new line. In figure 4, point design B is trended for various payloads ranging from 0 to 45 Mg, all with a 15 ft X 60 ft cargo bay. These plots are obtained by merely changing the value of the "fixed" masses in equation (10) by the amount of change in cargo mass. The constant performance line is shown for this dual fuel SSTO for a 50 X 100 n.m. orbit. This vehicle, if trended from a vehicle carrying 29.5 Mg to one carrying a payload of 45.4 Mg (100 Klb) would have a GLOW of approximately 2.44 Gg (5.4 Mlb) (figure 4, Pt. D).

In figure 5, these values are replotted for a constant orbit to show ascent propellant mass and vehicle gross mass as a function of payload carried. The growth factor from the slope of the curve is fairly constant at 24.5 to 1, for the mass range shown but closer inspection shows that payload is growing at a slightly faster rate than vehicle gross mass (and propellant loading).

In figure 6, sensitivities to changes in estimated fixed masses are shown. The most identifiable fixed masses, as stated earlier for these vehicles, are payload, payload bay and doors, crew compartment and avionics, totaling an estimated 45 Mg (100,000 lb).

However, from a plot in figure 6, it can be seen from the envelope of point designs as indicated by the ellipse for phase A/B orbiters that a fixed mass of 91 Mg (200 Klb) and an exponential of 1/C gives the best agreement for this type of vehicle. Winged orbiters from these two-stage systems are simply smaller single-stage-to-orbit (SSTO) vehicles, in that they have very similar propulsion, a cargo bay, a crew compartment, and internal LOX/LH_2.

Phase A/B boosters are shown on the trending curve, figure 6. They are essentially lower technology SSTO's with no cargo space and lighter thermal protection but with a turbojet cruise system not required by SSTO's, the SSTO vehicles having sufficient crossrange to "glide" back to base after return from orbit. It would not be unusual for a cruise-back system on a booster of a two-stage system to weigh well over 50 Mg (ref. 9 and 10) when the jet fuel is included.

Many factors dictate the sizing of a two-stage vehicle such as the design philosophy for the booster, whether heat sink and low staging velocity, or thermally protected and high staging velocity. But, it can be seen from figure 6 that a number of the phase A/B designs optimized at a point on the trending curves where increasing vehicle size further would yield diminishing rates of increase in performance (an expected solution).

From further inspection of figure 6, it can be seen that single-stage vehicles are located on the low slope portion of the curve where relatively large changes in vehicle size are required for small performance gains.

III. VEHICLE MASS ESTIMATING BY SUBSYSTEM TRENDING

Trending subsystem-by-subsystem is the technique used when a point design mass is to be established or when parametric-type analyses are to be made to study the effects of changes in one subsystem on the overall system. A sample computer case for this method has been included for the in-house, single-stage-to-orbit vehicle. (For computer printout, see Appendix B.) The method is described in detail in the following paragraphs. These paragraphs are numbered to correspond to the subsystem designations used in the widely accepted mass properties reporting system of reference 11. Groupings such as (a), (b), (c), etc., have been added for clarity.

(a) STRUCTURE AND THERMAL PROTECTION GROUP

1.0 WING

In reference 2, wing mass equations take into account landed mass, load factor, span, theoretical wing area, and root thickness. A revised equation is used which incorporates exposed wing and exposed span as opposed to total span and theoretical wing. Also, the wing carry-through is treated as a separate term. In addition, a wing efficiency factor is used to better reflect redistribution of total lift between body and wing as the relative size of wing and body change. The resulting wing equation is:

$$m_W = \left[N_Z \cdot m_L \quad \frac{1}{1 + f \dfrac{S_b}{S_W}} \right]^{n_1} \left[\frac{S_W}{T_r} \right]^{n_2} \left[W_m \left(L_W \right)^{n_2} + W_c \left(L_b \right)^{n_2} \right] \qquad (14)$$

In the equation above, m_L is landed mass, W_m and W_c are material configuration constants for wing and wing carry-through, respectively. Other symbols are defined in the symbols list. Earlier equations were structured from historical data for aircraft with relatively narrow bodies and distinct wing carry-throughs and typically were of the following form:

$$m_W = \left(N_Z \cdot m_L\right)^{n_1} \left[\frac{S_{THEO} \text{ X TOTAL SPAN}}{T_{Root}}\right]^{n_2} \qquad (15)$$

where: S_{THEO} = theoretical wing area

T_{Root} = theoretical wing root thickness

n_1 and n_2 = exponentials (see Table III)

By separating carry-through and exposed wing (last bracketed term) in equation (14), the equation is more flexible and more accurate for a wider range of Earth-to-orbit vehicles and does not give large errors when, for instance, the area ratio of body-to-wing is large. In the extreme, a vehicle with a wide body and small fins (for wings) would have a large theoretical wing but very small exposed wing. In this latter case, an equation (which is based solely on theoretical wing) gives large errors in wing mass.

Similar problems were found with the first (or loads) term of the older wing equation since the equation did not recognize the lift distribution between body and wing for various body-to-wing area ratios. To improve this aspect of the equation, a body/wing efficiency factor, f, has been added along with a body-to-exposed wing area ratio. The body wing efficiency factor applied to the ratio recognizes these changes in wing/body lift distribution due to body size.

In the proposed equation, when the exposed wing area, S_W, approaches zero or $S_b/S_W \rightarrow \infty$ wing mass approaches zero, a desired result. Conversely, as wing area becomes larger (i.e., body area small), the term S_b/S_W approaches zero and the first (bracketed) term in the proposed equation approaches a maximum value of $N_Z m_L$, which is the normal load factor times the landed mass of the entire vehicle. The vehicle is, in essence, a flying wing and all the vehicle load is carried by the wing. Both of the above changes in the wing equation, i.e., the modification to the loads or first bracketed term, and the wing and carry-through geometry, or last two terms, gives better agreement with known advanced SSTO point design vehicle wing masses. It does not give erroneous results when, as cited above, large changes are made in the relationship between wing and body planform areas.

Also included in equation (14) are wing and wing carry-through material/configuration constants which can be varied for a specific point design. All the wing-related constants are very sensitive to material selections, whether metallic, metallic composite, or organic composite; or to configuration, whether skin stringer or honeycomb. Other factors adding to the complexity of the selection is whether the wing is dry, wet (cryogenic or storables); whether the thermal protection system is integral with the wing or an add-on, therefore, not appearing in the wing weight statement but under thermal protection, or whether the wet wing is for a vertical takeoff or horizontal takeoff vehicle.

Equally complex is the wing carry-through which may be a separate structure (as in most aircraft) or integrated with body structure as in the shuttle and most of the SSTO's being studied.

Of equal importance to wing-mass trending are the methods used for comparison of unit structural masses. In the past, wing unit mass has been variously defined because of the variety of mass-to-area ratios used, namely for mass:

a) Exposed wing mass

b) Exposed wing plus wing carry-through

c) a) and b) with or without thermal protection

and for areas:

a) Exposed wing planform

b) Structural planform (exposed wing+structural wing carry-through)

c) Theoretical wing

To add to the complexity of the definitions, the effects of wing size on unit mass must be considered even when the mass-to-unit area ratio definition is agreed upon. From equation (14) and the definition of unit mass, as:

$$m_{w_u} = m_w/S_w \tag{16}$$

It can be shown using equation (14) that, for equal wing loadings, the relationship between unit masses for dimensionally different but geometrically similar wings is related by some exponential, usually not zero, or:

$$\frac{m_{w_{u_2}}}{m_{w_{u_1}}} = (F)^{n_3} \tag{17}$$

where: F is the ratio of dimensions of wing point designs, 2 and 1, and n_3 is some exponential. For the wing equation of reference 1, this value is 0.312 or the unit mass of the wing is growing roughly as the cube root of dimensional ratio even though wing loading is constant.

2.0 TAIL

Tail unit mass is also size dependent and the equations reflect this fact. One such equation for tail mass in reference 1 is:

$$m_t = V_t \left(S_t\right)^{1.24} \tag{18}$$

Like the wing, "V_t" the material configuration constant, is dependent on the type of materials and construction used. Defining tail unit mass as gross structural tail mass divided by area, or dividing both sides of equation (18) by S_t, and assuming area is a function of dimension squared for a photographically enlarged tail:

$$\frac{m_{t,u_2}}{m_{t,u_1}} = (F)^{0.48} \tag{19}$$

This is a similar result to that found in wing equations based on historical data. The in-house SSTO vehicle mentioned earlier has a tail profile area of 125.4 m^2 (1350 ft^2) while the shuttle has an area of only 38.4 m^2 (413 ft^2). The area ratio is 3.27 or dimensional ratio (assuming exactly similar geometry) is approximated by $(3.27)^{1/2}$ or size factor F = 1.81. From equation (19), the ratio in unit masses for the same technology level due to size effects alone is 1.33 or a 33 percent increase can be expected.

3.0 BODY GROUP

The body group is more difficult to assign to a mass equation because of the unique features of each generic design. For Level I masses, the major subelements included in the body masses are crew compartment, body shell, thrust structure and body flap. In earlier phase A/B shuttle studies (references 8 through 10), the main propellant tanks were carried in the main propulsion system mass unless they were load carrying. In this latter case, they were carried under body structure. For consistency in mass property reporting in this paper, they are always carried in the body group, whether load carrying or not.

The body mass equation recommended for Level I mass properties determination (when constants are selected for the generic design) is:

$$m_b = B_c\,(N_c)^{.5} + B_b \cdot S_{WET_b}\,(N_Z)^{1/3} + B_f\,(V_2) + B_o\,(V_1)^{1.1} + B_t \cdot T_{VAC_{N_e}} + B_{bf} \cdot S_f^{\;1.15} \qquad (20)$$

The above subelements are easily identifiable in most preliminary designs and enhance the accuracy of body mass and c.g. estimates as opposed to using an overall mass average for all body elements. In the above equation B_c, B_b, B_f, B_o, B_t, and B_{bf} are material configuration constants for cabin, body proper, main fuel tank, main oxidizer tank, thrust structure, and body flap. The number of crew, mission specialists or passengers, "N_c" appears in the body equation since the size of the pressurized cabin space is related to the number of persons to be accommodated. "N_Z" is found from the product of safety factor times an ultimate load factor or is 1.4 X 3.75 for the shuttle.

In the body equation, non-optimums should be accounted for in the "B_b" constant. These should include door cutouts, doors, gear wells, intertank adapters (between main propellant tanks), thermal protection system support fairings, ring frames and stringers (where required), cargo bay, and other miscellaneous structural subelements. The term S_{wet_b}, for the body should not include those areas of the body for which the integral tankage forms the moldline.

For a tank, assuming equal ullage pressure, P_u, and material allowables, σ, tank wall thickness from the hoop stress formula is:

$$t = \frac{P_u}{\sigma} \cdot r \tag{21}$$

Now tank surface area, assuming the same tank proportions, is given by:

$$S_{T_K} = (\text{Constant})\ r^2 \tag{22}$$

Tank mass is found from the product of area, thickness, and density of the material of which the tank is made, or:

$$m_T = A_T\ t\ \rho \tag{23}$$

In the above equation for tank mass, the area of the tank can be defined as a constant times tank radius and the value of tank w[illegible] thickness by equation (21). Equation (23) then becomes:

$$m_T = (\text{constant})\, r^2 \times \frac{P_u}{\sigma}\, r = (\text{constant})\, r^3 \tag{24}$$

Now tank volume

$$\text{Vol.} = \text{constant}\,(r^3) \tag{25}$$

Therefore, masses and volumes are both functions of r^3 or,

$$\frac{\text{Tank mass}}{\text{Tank volume}} = (\text{constant}) \tag{26}$$

The above statement is valid assuming uniform pressure throughout the tank with no insulation, no changes in non-optimums or tank loads with size. In a gravity field, or under the influence of engine thrust, however, tank hydraulic pressures increase as a function of tank dimension along the line of action of thrust or gravity axis. At the bottom of a tank containing a dense propellant, such as LOX, the hydraulic pressure alone at the bottom of a 60 foot tank under 1.3 g's acceleration is 297 kPa (43 psi) compared to the shuttle external LOX tank design ullage pressure of 262 kPa (38 psi). It can be seen that tank mass is, therefore, sensitive not only to tank wetted area and wall thickness due to ullage pressure, but is also affected by hydraulic head linearly increasing from the free surface of the fluid. Because of this, the mass center of tanks carrying relatively dense propellants falls aft of the tank centroid and further, the tank structural mass grows faster than a linear factor with volume. Because of hydraulic head, if convenient from other packaging considerations, it is desirable to utilize tanks for dense propellants such as RP and LOX, which are as short as possible along the resultant of the thrust and gravity axes. The hydraulic head effects of LH_2 are not considered significantly large and are ignored for purposes of trending. LH_2 insulation masses, however, are significant for a hydrogen tank and should be accounted for. Generally, it is found there is no payload advantage in insulating LOX tanks (ref. 12) but other factors may make this necessary.

Tank configuration effects are significant and some tanks are long and slender giving rise to higher hydraulic heads for tanks carrying relatively dense propellants such as LOX and RP. Tapered tanks are also heavier.

4.0 INDUCED ENVIRONMENTAL PROTECTION

In the past, induced environmental protection (when distinct and separate such as the shuttle system of silica tile) has often been based on a constant times wetted area. In this paper, this equation is being updated to reflect sensitivities to ballistic coefficient and thermal capacity of the backup structure, namely:

$$m_{TPS_u} = K_R \left[\frac{1}{\bar{t}}\right]^a \left[\frac{m_e}{S\ C_L}\right]^b \quad (S_{WET_v}) \qquad (27)$$

where K_R is the material/configuration constant for the thermal protection and $\bar{t}$ is a constant reflecting heat capacity of the backup structure; m_e is vehicle entry mass.

The equation is based on an assumption of equilibrium glide conditions and equal entry times for the point design and off-point design vehicle. If flow over the entire vehicle is laminar during the entire entry, the exponential "b" would be 0.5 for the assumptions made. On the other hand, if the flow over the entire vehicle is turbulent during the entire entry, the exponential would be 0.8. Because of the lack of current knowledge as to when transition occurs and over what percentage of the vehicle body, it is not possible to determine exact vehicle point design thermal protection masses. The equation is useful, however, in vehicle design when differing assumptions are made for entry planform loading, C_L, backup structure, and the amount of turbulent flow merely to establish trends in vehicle mass. The estimator studying such trends between one point design and another should be consistent in the exponential used; without more detailed information, an exponential of 0.65 would be a reasonable choice.

A value for the constant K_R was obtained for the reusable surface insulation concept by substituting the known quantities in equation (27) and solving for K_R for the shuttle design. The shuttle-derived constant is based on the assumption that flow is laminar over the vehicle. The constant does not include allowance for the mass of carrier panels. Many of the concepts being considered require carrier panels on the isogrid main body propulsion tanks but not, for instance, on a smooth wing stress skin.

5.0 LANDING, DOCKING, AND RECOVERY

For this category in the Level I weight statements, the landing gear mass is taken simply as a percentage of landed mass. For the shuttle, the manipulator and tank separation system are included in this category.

When these masses are deleted in the shuttle Mass Properties Report (ref. 7), the landing gear mass, as a percentage of landed mass, equals 3.3 percent.

If composites are extensively used, it is estimated that this figure can be reduced to 3.0 percent. If skids were operationally practical and adaptable to a large SSTO, the 3.0 percent figure could conceivably be reduced further to an estimated value of 2.55 percent. It is estimated that conventional gear without brakes would yield a similar figure to that shown for skids, i.e., 2.55 percent. (A ground arrester would be employed for stopping such a vehicle.) Separating the manipulator and docking functions is desirable since these items are mission oriented and not necessarily related to vehicle size. The equation for trending purposes then becomes:

$$m_g = K_L \times m_L + m_{ma} + m_S \qquad (28)$$

Where: K_L is the landing gear constant and m_{ma} and m_S are manipulator and separation systems masses, respectively; and m_L is the landed mass.

(b) PROPULSION GROUP

6.0 PROPULSION ASCENT

For sensitivity studies, once a "point design" main propulsion system is established, the total main propulsion system mass (not including main tankage) can be expressed as:

$$
m_{eng} = \left[R_{ph} + R_n (\varepsilon_1 - 1) + R_{ne} \frac{(\varepsilon_2 - \varepsilon_1)}{P_c} + R_{na} \frac{(\varepsilon_2^{1/2} - 1)}{(\dot{m} P_c)^{1/2}} + R_{pf} + R_{ga} I_{se} \right] \dot{m} N_e \tag{29}
$$

This equation is based on unpublished information obtained from LeRC. The terms in order of appearance in equation (29) refer to power head mass, fixed position nozzle mass, nozzle extension, extension actuator, pressurization and feed systems, and engine gimbal actuator mass. The latter is based on vacuum specific impulses. In the above equation, N_e is the number of engines in any generic grouping. This not only applies to type of fuel used, but also as to whether the engine is fixed or gimbaled, extendable or fixed nozzle. If the vehicle trending involves a fixed propulsion system design, then the bracketed term can be replaced with a constant and N_e or $\dot{m}$ varied with vehicle size for a constant or nearly constant thrust-to-weight.

7.0 PROPULSION - REACTION (OR ATTITUDE) CONTROL, RCS

The mass equation for the control (or attitude control) system used and devised here is based on the following assumptions:

1. That angular acceleration rate in pitch, yaw, and roll is directly proportional to applied torque and inversely as vehicle moment of inertia.

2. That applied torque, with only small variations, is proportional to vehicle size (e.g., double the physical size, or L_r, and moment arms from vehicle c.g. to RCS pods are roughly doubled).

3. That vehicle moment of inertia is proportional to the product of vehicle mass and radius of gyration or equals $m_e \times (e \times L_r)^2$ where "e" is a constant identifying radius of gyration location when multiplied by vehicle reference length, L_r.

When utilizing the above assumption and further assuming no change in the total RCS mission impulse required or angular rates:

$$m_R = R_{RCS} \times m_e \times L_r \tag{30}$$

A characteristic point design constant, R_{RCS}, for equation (30) for a storable system can be determined from the shuttle orbiter. Any significant change in an off-point design vehicle from the shuttle, such as the transfer of aft RCS modules from body to wing tips, substitution of cryogenic system for storable, change in engine thrust-to-vehicle mass ratio or total impulse requirement would require a redetermination of the values of the constant to avoid gross errors in mass estimation of the system. A cryogenic RCS system has a somewhat higher dry mass for the typical Earth-to-orbit transportation system, but the lower corrosivity and toxicity of these propellants may be the determining factors. For very high total impulse requirements, the cryogenic system, due to its higher specific impulse, has a clear-cut advantage in mass over the storable RCS system.

8.0 PROPULSION - ORBITAL MANEUVERING SYSTEM, OMS

Because of the mass advantage in propellant savings for the higher OMS impulse requirements over RCS, the cryogenic system in this case has a more clear-cut advantage. In the equations, engines and tanks are treated separately since, unlike the RCS system, OMS total ΔV requirements typically vary widely from mission to mission. The OMS engine mass including tanks and feed system mass is:

$$m_o = M_{me} \times N_e \times T_{vac} + M_t \times m_{op} \tag{31}$$

Where: M_{me} and M_t are constants for the maneuver engine, tankage and pressurization and feed, respectively.

(c) POWER GROUP

9.0 PRIME POWER

Both the auxiliary power unit and fuel cells are included in this category. The Auxiliary Power Unit is assumed to be designed by peak aerocontrol system requirements with engine gimbal requirements having a secondary effect. Constants shown in Table III are based on shuttle and the following surface control rates:

Elevon = 20^o/sec

Rudder = 14^o/sec

Body flap = -3^o/sec + 1^o/sec

Speedbrake (Priority rating logic)

For these rates, it is assumed that control surface power is directly proportional to the total aerocontrol movable surfaces exclusive of speedbrake. It is further assumed that, with reasonable accuracy, the control surface power constant, PW_c, in the equation below could be altered in direct proportion to the increase or decrease in control surface rates since power is proportional to rate. The Prime Power mass is then:

$$M_{pow} = PW_c \times S_c + PW_e \times \Sigma T_{VAC} + PW_b \times m_a \tag{32}$$

Where: PW_c, PW_e, and PW_b are constants reflecting demands from surface controls, main engines, and avionics subsystems, respectively.

Hydraulic power unit mass remains essentially constant with a pressure increase to 3.45×10^6 N/m^2 (5,000 psi); however, the mass of this subsystem will decrease with the utilization of accumulators to handle peak loads due to the reduction in total horsepower required and the more efficient operation at normal power loads of the unit (i.e., reduced peak to normal power ratios).

Prime power also includes battery power. For this subsystem, it is assumed that its mass is directly proportional to the mass of the avionics, m_a, for a given technology level, the last term in equation (32). A mass for an advanced technology avionics system is given in reference 6.

10.0 ELECTRICAL CONVERSION AND DISTRIBUTION

Electrical conversion and distribution is assumed to be proportional to vehicle landed mass or:

$$m_{e\ell} = E \times m_L \tag{33}$$

The above constant, E_r is assumed to be somewhat sensitive to mission but is relatively insensitive to configuration.

11.0 HYDRAULIC CONVERSION AND DISTRIBUTION

The mass trending equation for hydraulic conversion and distribution is similar to the prime power equation for hydraulics but with altered constants, namely:

$$m_h = H_{cs} \times S_c + H_e \times N_e \times T_{vac} \tag{34}$$

Where: H_{cs} and H_e are constants related to surface controls and engine gimbal actuation, respectively.

The baseline constants in Table III are derived from the shuttle. For advanced technology hydraulics (such as high pressure system), the two constants above are reduced due to the utilization of smaller diameter hydraulic lines.

12.0 SURFACE CONTROLS

Surface control mass (actuators, etc.), like the control power source, is assumed to be directly proportional to movable surface control area for a given rate, or:

$$m_{sc} = S_{sc} \times S_c + S_{pc} \tag{35}$$

In the above equation, S_c is surface control area and S_{sc} and S_{pc} are surface control and pilot control related constants. It is assumed that pilot related controls are independent of vehicle size.

(d) MISCELLANEOUS

13.0 AVIONICS

Avionics mass is assumed to be relatively insensitive to vehicle size and equals:

$$M_{av} \ (m_d)^{1/3}$$

Where: M_{av} is a constant and m_d is dry mass.

14.0 ENVIRONMENTAL CONTROL

Environmental control system is assumed to be sensitive to the wetted area of pressurized compartments or in terms of volume, an exponential of two-thirds. Pressurized compartments include wheel wells and cabin. (On the shuttle, the cargo bay is not pressurized.) For the electronics, it is assumed that heat input to the cabin, for a given technology, is directly proportional to electronics mass.

The total system mass for the cabin is relatively insensitive to mission duration. (The D factor shown below reflects principally oxygen container mass for the number of man-days on orbit,) or:

$$m_{ENV} = E_c \ (V_p)^{.66} + E_o \times N_c \ (D) + E_a \ m_a \qquad (36)$$

15.0 PERSONNEL PROVISIONS

This category includes the fixed life support system, food, waste, and water management systems, fire detection, pilot and crew stations. This category is relatively insensitive to mission duration beyond an estimated one-day limit. Below a one-day limit, it is assumed that the bulk of the food, waste, and water management systems could be removed, or that "PP_f" in the equation below would be reduced to zero. However, individual personnel provisions, such as seats, are directly related to the number of pilots, mission specialists, and passengers, and must be included. Therefore:

$$m_{pp} = PP_f + PP_s\ (N_c) \tag{37}$$

16.0 MARCIN (OR GROWTH ALLOWANCE)

Margin is equal to a constant times the dry mass of all the subsystems less engine mass. (NOTE: On the shuttle, a 10 percent growth margin is already included in the engine mass and the equation below is structured to be consistent with this practice.)

$$m_{margin} = MAR\ (M_d - N_e\ m_{eng}) \tag{38}$$

17.0 PERSONNEL

This categroy includes mass of crew, mission specialists, etc., and personnel-related GFE equipment and accessories, or:

$$m_{per} = P_m + P_p \times N_c \tag{39}$$

The above equation is applicable to one or more crew and is zero for an unmanned vehicle.

18.0 PAYLOAD PROVISIONS

Payload provisions are a fixed input to allow for any special installation and mounting equipment.

(e) PAYLOAD

19.0 PAYLOAD RETURNED

Payload returned is a fixed input.

(f) FLUIDS INVENTORY (ON ORBIT AND ENTRY)

20.0 RESIDUAL AND UNUSABLE FLUIDS

Residual and unusable fluids include gaseous propellants and pressurization gases in addition to trapped propellants. The amount of unusable fluids depends on manifold, line and sump design. For a given vehicle the residual or unusable fluids is taken as a constant times the ascent propellant mass to an exponential or:

$$m_{uf} = R_{rf} \times m_p^{.79} \tag{40}$$

21.0 RESERVES OMS AND RCS

Orbit maneuvering system and attitude control system reserves are determined by:

$$m_{ROMS} = m_L \left[e^{\left(\frac{R_o \, \Delta V_o}{I_{s_m} \, g} \right)} - 1 \right] \tag{41}$$

$$m_{RRCS} = m_L \left[e^{\left(\frac{R_r \, \Delta V_{Ro}}{I_{s_{ro}} \, g}\right)} - 1 \right] \qquad (42)$$

Where: R_o and R_r are constant percentages of ΔV requirements.

An overall constant may be substituted for the bracketed terms in equations (41) and (42) providing there are no changes in mission or engine design. The same is true for equations (43) and (44) in the next two sections.

22.0 RCS PROPELLANT ENTRY

Based on estimated reentry attitude control requirements of the shuttle and interpreted as an equivalent ΔV:

$$m_{RCS_e} = m_e \left[e^{\left(\frac{\Delta V_{Re}}{I_{s_{re}} \, g}\right)} - 1 \right] \qquad (43)$$

The specific impulse in the above equation is a degraded value reflecting a lower performance due to increased ambient pressure during entry. Entry mass is the vehicle mass after depletion of all usable RCS propellants.

23.0 RCS AND OMS PROPELLANT CONSUMABLES

On-orbit and deorbit attitude control and maneuver propellants are:

$$m_c = m_o + m_r$$

or:

$$m_c = m_{DESC} \left[e^{\left(\frac{\Delta V_o}{I_{s_m} g} \right)} - 1 \right] + m_{DESC} \left[e^{\left(\frac{\Delta V_{Ro}}{I_{s_{ro}} g} \right)} - 1 \right] \qquad (44)$$

Both the RCS and OMS requirements, it should be noted, are based on the vehicle mass at descent, which is defined herein as the vehicle mass with return cargo, entry RCS, and all residuals and reserves. In so doing, on-orbit maneuvers are assumed to take place after discharge of cargo. This is done for consistency in vehicle sizing but, of course, is sensitive to mission details.

(g) PAYLOAD DELIVERED

24.0 CARGO DISCHARGED

On-orbit net cargo or cargo delivered less cargo returned.

(h) FLUIDS INVENTORY (ASCENT PHASE)

25.0 ASCENT RESERVES AND ASCENT RESIDUALS

$$m_{ARES} = m_{INJ} \left[e^{\left(\frac{R_{ar}\, \Delta V_{IDEAL}}{I_{s_e} g} \right)} - 1 \right] + R_{ap} \cdot m_p \qquad (45)$$

The reserve propellant requirements are taken as a fraction of the ascent ideal ΔV to allow for launch dispersions. In addition, a percentage of the ascent propellant must be allowed for residuals.

Injected mass is gross less start-up and ascent propellants, inflight losses, ascent residuals and reserves; the ascent residuals and reserves being dumped prior to on-orbit maneuvers. A constant may also be substituted here for the bracketed term with reasonable accuracy providing trajectory and vehicle drag characteristics remain reasonably constant.

26.0 INFLIGHT LOSSES

Inflight losses include boiloff, prime power, environmental control, and hydraulic effluents or for mass estimating purposes:

$$m_{INF} = R_{INF} \times m_p \tag{46}$$

For very accurate trajectory analysis, the effluents can be reflected in fractionally lower specific impulse but it is considered sufficiently accurate to subtract one half the losses from the vehicle mass after depletion of usable ascent propellant and prelaunch losses.

27.0 ASCENT PROPELLANT

Ascent propellant is taken as usable propellant, or as the propellant load less prelaunch boiloff, preignition and thrust buildup losses or:

$$m_p = m \ (1 - R_{pl}) \tag{47}$$

Where: R_{pl} is a small fractional percentage (.001 to .002) and for system sizing purposes is inconsequential and, if not accounted for, merely results in a fractional percent error in the interpretation of ullage volume.

IV. GENERAL DISCUSSION

In the previous paragraphs, mass estimating relationships have been discussed in the order in which they appear in the mil standard 38310 (ref. 11). The trending equations have been structured to give better results for Earth-to-orbit transports. In addition, equation format has been improved over earlier equations which had previously been utilized primarily for commercial and military aircraft. Suggested constants have been provided in Table III.

Of all the subsystems, body structure is one of the heavier elements and on the shuttle, constitutes approximately 27 percent of the total dry mass. Similarly, this structural group is probably one of the most variable in total mass being heavily dependent upon configuration factors. One configuration factor is the cargo bay shape and location. For a single-stage-to-orbit winged vehicle, the cargo bay could be located in the nose of the vehicle and the main propellant tanks designed to form a simple body of revolution, one of the lightest possible structural masses could result. For aerodynamic and other mission oriented reasons, this may not be practical and various other shapes evolve. If cargo return is required from orbit, it may be necessary to locate the cargo bay in the vicinity of the vehicle normal c.g. to minimize "cargo-in" and "cargo-out" c.g excursions during horizontal flight. This latter location generally will effect a decrease in propellant tankage packaging efficiency and an increase in body structural non-optimums.

For the above reasons, no one set of constants can be provided which will apply to all body structural designs. The same is true, but to a lesser extent, of all subsystems. It is, therefore, up to the estimator to alter constants utilizing available information developed in structural analyses. In Table IV, suggested constants are given for main propellant tankage. If no detailed information is available, body mass can be structured from the tank mass estimating relationships of this table combined with estimates for intertank adaptors, fairings, cargo bay structure, non-optimums, and the other subelements suggested by equation (20) in the text. Once established for one size of vehicle body structure masses of the same generic design of other sizes can be obtained.

While the method of reporting mass properties follows closely that recommended for subsystems and fluids inventory in reference 11, it has been modified slightly so that it can serve both for Level I weight estimates of subsystems and also for sequential mass estimates. The sequential mass statement accounts for expendables during the mission such as main rocket and reaction control propellant and discharged cargo, etc. In this regard, item 19, Table III, has been changed to "cargo returned" while item (24), net "cargo discharged" has been added. In the more common type of missions, cargo up or down either remains the same, or decreases; and sequentially, vehicle mass is continually decreasing from liftoff to landing. However, if ascent cargo is less than return cargo, (such as on a retrieve mission), the cargo "discharged" on-orbit, item (24), should be listed as a negative quantity. A separate column can be used adjacent to individual subsystem masses for sequential masses. This mass generally showing a decreasing value from gross liftoff to dry mass as stated above.

V. EXAMPLE STRUCTURE

Example structure is shown for in-house study vehicle EN 155 in Appendix D. Composites and honeycomb were extensively used to obtain low mass. In Appendix E, unit masses of various types of crossections are shown. The sections apply to areas where exterior peak temperatures are approximately 1000^{o} C (1800^{o} F), and liquid hydrogen is stored internally. Two crossections (E-4 and E-5) however, are representative of areas where no fluid containment is required. The unit masses shown tend to be considerably lower than real structure since door cutouts, secondary structure, and other non-optimums are not included. The final figure in this section shows TPS unit masses as a function of peak temperatures for both metallic and ceramic reusable surface insulation. This data could be applied to mass estimation of the TPS on a vehicle for which isotherms for peak temperatures are known as opposed to use of the more general equation (27) in the text.

CONCLUSIONS

Several techniques for estimating the mass properties of Earth-to-orbit transportation systems have been reviewed. Where appropriate, basic equations and related constants have been revised, and estimating procedures given. Based on in-house studies, the following conclusions are offered:

1. The overall vehicle trending technique presented is considered very rapid for resizing a vehicle for which propellant and overall inert mass are already known and no significant configuration changes are involved in trending from one generic-point design to another.

2. Analysis on a more detailed basis by Subsystem Trending as outlined in this report has been found to be very accurate but requires more time and detailed information on individual subsystems.

3. The equations developed for subsystem trending are considered useful for parametric studies wherein the impact of one subsystem parameter on the overall system is being evaluated.

4. No one set of constants can be provided which will be suitable for each design. Constants provided are reasonable baselines from which more mature mass estimates can be developed when more detailed subsystem information is available.

INTERNATIONAL SYSTEM OF UNITS CONVERSION FACTORS,

PHYSICAL CONSTANTS, AND PREFIXES

(a) Conversion factors

CONVERT FROM	TO	MULTIPLY BY
INCHES	METERS	.025 400
$INCHES^2$	$METERS^2$	.000 645 160
FOOT	METERS	.304 800
$FOOT^2$	$METERS^2$	.092 903 040
POUNDS	KILOGRAMS	.453 592 370
$POUNDS/INCHES^3$	$KILOGRAM/METER^3$	27 679.905 . . .
$POUNDS/FOOT^3$	$KILOGRAM/METER^3$	16.018 463 . . .
FOOT/SEC	METER/SECOND	0.3048
$FOOT/SEC^2$	$METER/SECOND^2$	0.3048
$POUNDS/INCH^2$	$NEWTON/METER^2$ (OR PASCALS)	6894.757
$POUNDS/FOOT^2$	$NEWTON/METER^2$ (OR PASCALS)	14.788

(b) Prefixes

The names of multiples and submultiples of SI units may be formed by application of the prefixes:

FACTOR BY WHICH UNIT IS MULTIPLIED	PREFIX	SYMBOL
10^9	giga	G
10^6	mega	M
10^3	kilo	k
10^2	hecto	h
10	deka	da
10^{-1}	deci	d
10^{-2}	centi	c
10^{-3}	milli	m

(c) Physical constants

$g = 9.80665$ m/sec^2 or 32.174 Feet/Sec^2

REFERENCES

1. Anon.: Space Shuttle Synthesis Program, Vol. II., NASA CR-114987. December 1970.

2. Anon.: Handbook for Weight Estimating and Forecasting of Manned Space Systems During Conceptual Design, Vol. II., NASA CR-138536. November 1970.

3. Garrison, J. M.: Development of a Weight/Sizing Design Sythesis Computer Program, Vol. I., NASA CR-128867. February 1973.

4. Norton, P. J. and C. R. Glatt: VAMP: A Computer Program for Calculating Volume, Area, and Mass Properties of Aerospace Vehicles. NASA CR-2419. September 1974.

5. Eldred, C. H. and Gordon, S. V.: A Rapid Method for Optimization of the Rocket Propulsion System for Single-Stage-to-Orbit Vehicles. NASA TN D-8078. July 1976.

6. Anon.: Technology Requirements for Advanced Earth Orbital Transportation Systems. NASA CR-2879. December 1977.

7. Anon.: Space Shuttle Orbiter Mass Properties Status Report. CR-150952. May 2, 1976.

8. Anon.: Space Shuttle Phase B Final Report, Vol II., Technical Summary, Book 2, Orbiter Definition. NASA CR-119776. June 1971.

9. Anon.: Space Shuttle System, Part II(B) Orbiter Details. Detail Mass Properties Report prepared by MDAC under NAS 8-26016. NASA CR-119880. June 1971

10. Anon.: Space Shuttle Final Technical Report, Vol. II. Final Vehicle Configurations. Prepared by GDC under NAS 9-9207. NASA CR-102550. October 1969.

11. MIL-M-38310B (USAF): Mass Properties Control Requirements for Missile and Space Vehicles, Amendment 2, January 1976.

12. Kline, R. L. and Mendelsohn, A. R.: Thermal Integration Consideration for the Space Shuttle. Paper contributed by Grumman Corp. for presentation to ASME Space Technology and Heat Transfer Conference. Los Angeles, CA. June 21-24, 1970.

13. Glatt, C. R.: WAATS - A Computer Program for Weights Analysis of Advanced Transportation Systems. NASA CR-2420. September 1974.

14. **Bohon, H. L., Shideler, J. L., and Rummler, D. R. "Radiative Metallic Thermal Protection Systems: A Status Report" Journal of Spacecraft and Rockets, Vol. 14, No. 10, pp. 626-631, October 1977.**

TABLE I

TRENDING SUBSYSTEM GROWTH VERIFICATION

SUBSYSTEM	MASS RELATED TO	APPROXIMATE VEH. REF. LENGTH PROPORTIONALLY	APPROX. % OF TOTAL MASS
1.0 WING GROUP	PLANFORM AREA	L^2	10
2.0 TAIL	PLANFORM AREA	L^2	2
3.0 BODY	BODY WETTED AREA	L^2	26
TANKAGE	TANK VOLUME	L^3	16.5
THRUST STR.	THRUST	L^3	1.0
4.0 TPS	WETTED AREA	L^2	
5.0 LANDING GEAR	LANDED MASS	L^2	3
6.0 PROPULSION ASCENT	GROSS MASS	L^3	22.6
8.0 PROPUL. AUX. (INCL. PROP.)	INJECTED MASS	L^2	1.7
9.0 PRIME POWER	CONTROL SURFACE AREA	L^2	"
10.0 ELEC. POWER		L^2	"
11.0 HYDRAULICS	CONTROL SURFACE AREA	L^2	1.7
12.0 SURFACE CONTROLS	AREA	L^2	"
13.0 AVIONICS	MISSION DEPENDENT	CONSTANT	"
14.0 ENVIRONMENTAL CONTROL	FUNCTION OF PRESSURIZED TOTAL VOLUMES	CONSTANT	"
15.0 PERSONNEL PROVISIONS	SIZE OF CREW	CONSTANT	"
18.0 GROWTH	CONSTANT x DRY WT.	CONSTANT	"

TABLE II

SEMP PROGRAM VERIFICATION

(SHUTTLE ORBITER)

LEVEL I - WEIGHT STATEMENT

	R.I. 12/76	SEMP PROGRAM (11/4/77)
SUBSYSTEM	WT, LB	WT, LB
1.0 WING GROUP	15,098	15,657
2.0 TAIL GROUP	2,848	2,911
3.0 BODY GROUP	42,941	41,961
4.0 TPS	21,187	24,380
5.0 LANDING	7,713	8,041
6.0 PROPULSION	28,234	28,200
7.0 PROPULSION, RCS	2,755	2,814
8.0 PROPULSION, OMS	2,899	2,976
9.0 PRIME POWER	2,999	3,030
10.0 ELEC CONV AND DISTRIBUTION	7,133	7,310
11.0 HYDRAULICS	1,855	1,755
12.0 SURFACE CONTROLS	2,688	2,615
13.0 AVIONICS	5,946	6,011
14.0 ENVIRONMENTAL CONTROL	5,333	5,270
15.0 PERSONNEL PROVISIONS*	1,483	1,021
16.0 MARGIN	767	767
DRY WEIGHT	151,879	154,739

P/L PROVISIONS (467 LB) ARE INCLUDED IN PERSONNEL PROVISIONS **BY ROCKWELL INTERNATIONAL, NASA ORBITER PRIME CONTRACTOR.**

	R.I. 12/76	SEMP PROGRAM
	WT, LB	WT, LB
DRY WEIGHT	151,879	154,739
17.0 PERSONNEL	2,644	2,640
18.0 PAYLOAD ACCOMMODATIONS	1,608	1,608
19.0 CARGO (RETURNED)	32,000	32,000
20.0 RESIDUAL FLUIDS	1,523	1,551
LANDED WEIGHT	189,654	192,538
21.0 OMS AND RCS RESERVES		77
ENTRY WEIGHT	189,654	192,615
22.0 RCS PROPELLANT (ENTRY)		828
DESCENT WEIGHT	189,654	193,444
23.0 ACPS CONSUMABLES (RCS + OMS) ON ORBIT — RCS	5,909	1,664
OMS	16,304	12,883
24.0 CARGO DISCHARGED	33,000	33,000
INJECTED WEIGHT	244,867	240,990
25.0 ASCENT RESERVES AND ASCENT PROPELLANT RESIDUALS	2,344	4,454
26.0 INFLIGHT LOSSES	2,753	662
27.0 ASCENT PROPELLANT	5,206	5,206
GROSS LIFTOFF WEIGHT	255,170	251,313

TABLE III - VEHICLE SUBSYSTEM TRENDING

SUBSYSTEM	EQUATIONS AND VARIABLES	EQUATION CONSTANTS
1.0 WING GROUP	$m_w = \left[N_z \cdot m_L \dfrac{1}{1+f\dfrac{S_b}{S_w}} \right]^{.386} \left(\dfrac{S_w}{T_r} \right)^{.572} \left[W_m \cdot L_w^{.572} + W_c \cdot L_b^{.572} \right]$ WHERE: N_z = Ultimate normal load factor for the design condition (or 1.4 X 2.5 g's subsonic maneuver when ascent wing loading is limited to the subsonic maneuver case. m_L = Mass of vehicle at landing. S_b = Body planform area S_w = Exposed wing planform T_r = Exposed wing root chord max. thickness L_w = Exposed total structural wing span L_b = Body width at wing body juncture	Exposed wing material/configuration constants W_m = 0.286- Aluminum skin/stringer, dry wing, no TPS = 0.343- same as above but wet wing for storable propellant. = 0.229- metallic composite (Boron Aluminum) honeycomb dry wing, no TPS. = 0.263- same as above but wet wing for storable propellant such as RP = 0.214- Organic composite honeycomb, no TPS. = 0.453- Honeycomb dry wing super alloy hot structure no TPS required. Wing Carry - Thru Constants W_c = 0.0267 dry carry-thru (integral) = 0.0347 wet carry-thru (integral) = 0.100 dry carry-thru (conven.) = .12 wet carry-thru (conven.) Wing/Body efficiency factor f = 0.20 for conventional vehicle to 0.15 for control configured vehicle.
2.0 TAIL GROUP	$m_t = V_t (S_t)^{1.24}$ WHERE: S_t = tail profile area	Tail material/configuration constant V_t = 1.872 - aluminum skin/stringer, no TPS. = 1.108 - metallic composite structure, no TPS. = 1.000 - graphite epoxy composite structure, no TPS

TABLE III (CONTINUED)

SUBSYSTEM	EQUATIONS AND VARIABLES	EQUATION CONSTANTS
2.0 TAIL GROUP (CONT'D)		= 1.500 - super alloy honeycomb, hot structure.
3.0 BODY GROUP	$m_b = B_c (N_c)^{.5} + B_b \cdot S_{wet_b} \cdot (N_z)^{1/3} + B_f \cdot V_2 + B_o \cdot V_1^{1.1} + B_t \left[N_{e1} T_{vac1} + N_{e2} T_{vac2}\right] + B_{bf} \cdot S_f^{1.15}$	CABIN CONSTANTS
	WHERE:	B_c = 2043 - full windshield aluminum construction.
	N_c = number of crew	= 1293 - above with no windshield
	N_z = ultimate normal load factor (same as for wing)	= 1740 - full windshield composite construction.
	S_{wet_b} = wetted area of body structure less areas of main propellant tankage which dual as body shell	= 1140 - above with no windshield and composite construction.
	V_2 = volume of main fuel tank	BODY CONSTANTS
	V_1 = volume of main oxidizer tank	B_b = 2.72 - composite structure, no TPS
	N_e = number of engines (Type 1 or Type 2)	= 3.20 - aluminum structure, no composites, no TPS
	T_{vac} = engine vacuum thrust	= 3.40 - hot metallic Ti/Rene HC, no TPS required
	S_f = body flap planform area	= 4.43 - moldline tankage; tank, body structure, cryogenic insulation integrated.
	THRUST STRUCTURE CONSTANTS	TANK CONSTANTS
	B_t = .0030 Alum	B_f = fuel tank constant (see table IV).
	= .0024 Composites	B_o = oxidizer tank constant (see table IV).
		BODY FLAP CONSTANTS
		B_{bf} = 1.59 - hot structure
		= 1.38 aluminum str. no TPS

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TABLE III (CONTINUED)

SUBSYSTEMS	EQUATIONS	EQUATION CONSTANTS
4.0 THERMAL PROTECTION SYSTEM	$m_{TPS} = K_r \left[\frac{1}{\bar{t}}\right]^a \left[\frac{m_e}{SC_L}\right]^b S_{wet_V}$ WHERE: K_r = material/configuration constant for the TPS $\bar{t}$ = equivalent thermal thickness of backup structure (inches) m_e = vehicle entry mass S = Vehicle total entry planform area C_L = vehicle average lift coefficient during peak heating S_{wet_V} = vehicle wetted area $a = 0.302$ $b = 0.5$ laminar flow $b = 0.8$ turbulent flow	RSI MATERIAL CONSTANT K_r = .140 RSI (shuttle technology)* = .110 RSI Advanced* = .145 metallic BACKUP STRUCTURE CONSTANTS EQUIVALENT THICKNESS $\bar{t}$ = .100 aluminum skin stringer = .085 Titanium = .115 Graphite/Epoxy ENTRY TRAJECTORY C_L = .65 average to mach 10 (shuttle VTO/RSI) = .52 average to mach 10 horizontal take-off metallic TPS (BASIS FOR SHUTTLE $\frac{m_e}{S}$ = 49.9 LB/FT^2)
5.0 LANDING GEAR	$m_g = K_L \times m_L + m_{ma} + m_s$ WHERE: m_L = landed mass m_{ma} = manipulator mass (For shuttle = 805 lb) m_s = separation system mass	K_L = a constant percentage of landed mass for landing gear = .0330 shuttle gear = .0300 advanced composite gear = .0255 composite skid system, or composite wheel system with no brakes
6.0 PROPULSION	$M_{eng} = \left[R_{ph} + R_n (\varepsilon_1 - 1) + R_{ne} \frac{(\varepsilon_2 - \varepsilon_1)}{P_c} + R_{na} \frac{(\varepsilon_2^{1/2} - 1)}{(\dot{m} P_c)^{1/2}} + R_{pf} + R_{ga} I_{se} \right] N_e \dot{m}$	

*DOES NOT INCLUDE CARRIER PANELS

TABLE III (CONTINUED)

SUBSYSTEMS	EQUATIONS	EQUATION CONSTANTS
6.0 PROPULSION (CONT'D)	WHERE:	POWER HEAD CONSTANTS
	m_{eng} = total mass of all generic engine groupings	R_{ph} = sec, power head mass constant
	ϵ_1 = expansion ratio of first expansion	= 5.34 LOX/LH_2, P_c = 3000 psi
	ϵ_2 = expansion ratio of second expansion (when applicable)	= 5.18 dual fuel engine, P_c = 3000psi
	P_c = chamber pressure	= 2.48 LOX/hydrocarbon staged combustion P_c = 4000psi
	$\dot{m}$ = mass flow per engine	= 2.10 LOX/hydrocarbon LH_2 generator, P_c = 4000psi
	N_e = number of engines of any one generic grouping	NOZZLES
	I_{se} = vacuum specific impulse (nozzle extended for a two position engine)	R_n = sec, basic nozzle mass constant
		= .01194 LOX/LH_2
		= .00727 LOX/hydrocarbon
		= .015 EN 155 (dual fuel)
		NOZZLE EXTENSIONS
		R_{ne} = sec, nozzle extension mass constant
		= 9.943 LOX/LH_2
		= 6.054 LOX/HC
		NOZZLE EXTENSION ACTUATION
		R_{na} = sec^{½}/in. constant for extension mechanism
		= 60.54 LOX/LH_2
		= 36.86 LOX/HC

TABLE III (CONTINUED)

SUBSYSTEMS	EQUATIONS	EQUATION CONSTANTS
6.0 PROPULSION (CONT'D)		PRESSURIZATION AND FEED R_{pf} = lines manifold and pressurization system = 1.64 current technology = 1.4 Composite/metallic feedlines GIMBAL ACTUATORS R_{ga} = .00129
7.0 PROPULSION RCS (ATTITUDE CONTROL SYSTEM)	$m_r = R_{RCS} \times M_e \times L_r$ WHERE: m_e = entry mass L_r = vehicle reference length	R_{RCS} = point design all-up system constant including tanks, pressurization and feed, gimbal actuator, etc. = 1.36×10^{-4}/FT. based on the shuttle storable system = 1.51×10^{-4}/FT based on a cryogenic system
8.0 PROPULSION, OMS (ORBITAL MANEUVER SYSTEM)	$m_o = M_{me} \times N_e \times T_{vac} + M_t \times m_{op}$ WHERE: N_e = number of maneuver engines T_{vac} = vacuum thrust per maneuver engine m_{op} = required maneuver propellant	M_{me} = maneuver engine constant LBT/LBm = .0863 based on current shuttle storable system = .035 based on advanced cryogenic space engine M_t = maneuver system propellant supply system = .119 for storable propellants including pressurization = .152 for cryogenic propellants including pressurization and feed

TABLE III (CONTINUED)

SUBSYSTEMS	EQUATIONS	EQUATION CONSTANTS
9.0 PRIME POWER	$m_{pow} = PW_c \times S_c + PW_e \times \Sigma T_{vac} + PW_b \times m_a$ S_c = total surface control area ΣT_{vac} = total vacuum thrust of gimbaled engines m_a = mass of avionics	PW_c = surface control hydraulic pump power demand = .712 = .610 (accumulators for peak demand) PW_e = Engine gimbal power demand = $.97 \times 10^{-4}$ PW_b = battery power demand constant = .405
10.0 ELEC. CONV. AND DISTR.	$m_{e\ell} = E \times m_L$ WHERE: $m_{e\ell}$ = mass of electrical system m_L = vehicle landed mass	E = electrical conversion and distribution system mass constant = .02 = .038 (SHUTTLE)
11.0 HYDRAULICS CONVERSION AND DISTRIBUTION	$m_h = H_{cs} \times S_c + H_e \times N_e \times T_{vac}$ WHERE: S_c = total surface control area ΣT_{vac} = total vacuum thrust of gimbaled engines	H_{cs} = surface control constant = 2.10 shuttle technology base = 1.23 for a 5000 psi system H_e = engine related gimbal actuation = 3.00×10^{-4} shuttle technology = 1.68×10^{-4} for a 5000 psi system
12.0 SURFACE CONTROLS	$m_{sc} = S_{sc} \times S_c + S_{pc}$ WHERE S_c = surface control area	S_{sc} = surface control actuators constant = 3.75 for shuttle technology

TABLE III (CONTINUED)

SUBSYSTEMS	EQUATIONS	EQUATION CONSTANTS
12.0 SURFACE CONTROLS (CONT'D)		3.80 = for 5000 psi system 3.32 = for 5000 psi system of advanced materials S_{pc} = miscellaneous systems = 200
13.0 AVIONICS	$m_{av} = M_{av} (m_d)^{1/8}$	M_{av} = avionics mass constant = 1350 for current technology = 710 for 1990 technology
14.0 ENVIRONMENTAL CONTROL	$m_{env} = E_c (V_p)^{.75} + E_o \times N_c (D) + E_a m_{av}$ WHERE: V_p = total pressurized volume including wheel wells N_c = number of crew D = days on orbit m_{av} = avionics mass	E_c = pressurized volume constant = 5.85 E_o = oxygen supply tank constant = 10.9 E_a = avionics heat load constant = .44
15.0 PERSONNEL PROVISIONS	$m_{pp} = PP_f + PP_s (N_c)$ WHERE: N_c = number of crew (1 to 4)	PP_f = food waste and water management system, 1 to 4 crew. = 0 for mission $\leq$24 hours = 353 for missions $\geq$24 hours PP_s = seats and other pilot and crew related items = 167

TABLE III (CONTINUED)

SUBSYSTEMS	EQUATIONS	EQUATION CONSTANTS
16.0 MARGIN	$m_{mar} = MAR\,(m_d - \Sigma N_e\, m_{eng})$ m_d = vehicle dry mass m_{eng} = engine mass N_e = number of engines	$MAR = 0.1$
17.0 PERSONNEL	$m_{per} = P_m + P_p\,(N_c)$ where: N_c = no. of crew (1 to 4)	P_m = miscellaneous = 400 P_p = personnel = 560
18.0 PAYLOAD ACCOM.	Fixed Input	
19.0 CARGO (RETURNED)	Fixed Input	
20.0 RESIDUAL FLUIDS	$m_{uf} = R_{rf}\,(m_p)^{.79}$	$R_{rf} = .05$ (includes main propellant tank pressurization gas)
21.0 OMS AND RCS RESERVES	$m_{ORR} = m_L \left[e^{\left(\frac{R_o\, \Delta V_o}{I_{S_m}\, g}\right)} + e^{\left(\frac{R_r\, \Delta V_{re}}{I_{S_{ro}}\, g}\right)} - 2 \right]$ where: ΔV_o & ΔV_{ro} = equivalent OMS and RCS ΔV's for the mission.	$R_o = .005$ $R_r = .005$ $g = 32.2$ ft/sec^2 for specific impulse values, see (23)

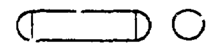
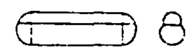
TABLE III (CONTINUED)

	SUBSYSTEMS	EQUATIONS	EQUATION CONSTANTS
22.0	RCS PROPELLANT (ENTRY)	$m_{RCS_e} = m_e \left[e^{\left(\frac{\Delta V_{re}}{I_{s_{re}}\, g}\right)} -1 \right]$ where: m_e = vehicle entry mass ΔV_{re} = "Delta Vee" equivalent for entry = 40 ft/sec for shuttle	$I_{s_{re}}$ = effective entry avg. specific impulse; = 242 sec. for shuttle including degradation for back pressure g = gravity constant
23.0	ACPS CONSUMABLES (RCS + OMS) ON ORBIT	$m_c = m_{DESC} \left[e^{\left(\frac{\Delta V_{ro}}{I_{s_m}\, g} + \frac{\Delta V_o}{I_{s_{ro}}}\right)} -2 \right]$ where: m_c = consumable RCS + OMS ΔV_o = equivalent ideal ΔV for maneuver I_{s_m} = average specific impulse of maneuver engine with three restarts $I_{s_{ro}}$ = average specific impulse (pulsing)	I_{s_m} = 313 sec. (storable) I_{s_m} = 440 sec. (cryo) $I_{s_{ro}}$ = 289 sec. (storable pulsing) $I_{s_{ro}}$ = 398 sec. (cryo pulsing)
24.0	CARGO DISCHARGED		Ascent cargo less return cargo
25.0	ASCENT RESERVES AND ASCENT PROPELLANT RESIDUALS	$m_{ARES} = m_{INJ} \left[e^{\left(\frac{\Delta V_{IDEAL}\, R_{ar}}{I_{s_e}\, g}\right)} -1 \right] + R_{ap}\,(m_p)$ ΔV_{IDEAL} = ideal I_{s_e} = main engine vacuum specific impuls g = gravity constant	R_{ar} = .005 ascent reserves constant R_{ap} = .004 ascent residuals constant m_p = ascent propellant
26.0	INFLIGHT LOSSES	$m_{inf} = R_{inf}\,(m_p)$	R_{inf} = .0043 inflight losses constant

TABLE III (CONCLUDED)

SUBSYSTEMS	EQUATIONS	EQUATION CONSTANTS
27.0 ASCENT PROPELLANT	$m_p = m\ (1 - R_{PL})$	R_{PL} = .001 to .002 for pre-launch losses; m = total propellant

TABLE IV - TANK WEIGHT CONSTANTS

SOURCE	PROPELLANT	K LB/FT^3	TANK DESCRIPTION					
			VOL FT^3	ULLAGE PRESSURE	INTEGRAL OR NON-INTEGRAL	MATERIAL	GEOMETRY	COMMENTS
SHUTTLE E/T	LH_2	.5918	53,515	36	INTEGRAL	AL2219		DOES NOT INCLUDE INSULATION
EN-155*	LH_2	.8430	60,?37	30	INTEGRAL	INC 718		HONEYCOMB SANDWICH ADDED HONEYCOMB FOR THERMAL PROTECTION
EN-178*	LH_2	.5760	41,646	20	INTEGRAL	AL2219		ISOGRID INCLUDES 4,364 LB INSULATION
SHUTTLE E/T	LOX	.6458	19,609	38	INTEGRAL	AL2219		DOES NOT INCLUDE INSULATION
EN-155*	LOX	.7660	18,355	20	NON INTEGRAL	AL2219		POLYMIDE HONEYCOMB FOR INSULATION AND STRUCTURAL STABILIZATION
EN-178	LOX	.5160	21,841	15	INTEGRAL	AL2219		ISOGRID INCLUDES 1,704 LB INSULATION
S-1C	LOX	.804	47,250		INTEGRAL	AL2219		
EN-155	JP-5	.7000	4,819	5	NON INTEGRAL	AL2219		CONVENTIONAL SKIN STIFFENED CONSTRUCTION W/O INSULATION
	JP-5	.28					N/A	PENALTY FOR DRY-WET WING
S-1C	RP-1	.867	30,000		INTEGRAL	AL2219		

*NOTE: EN DESIGNATIONS REFER TO LARC IN-HOUSE STUDY VEHICLES.

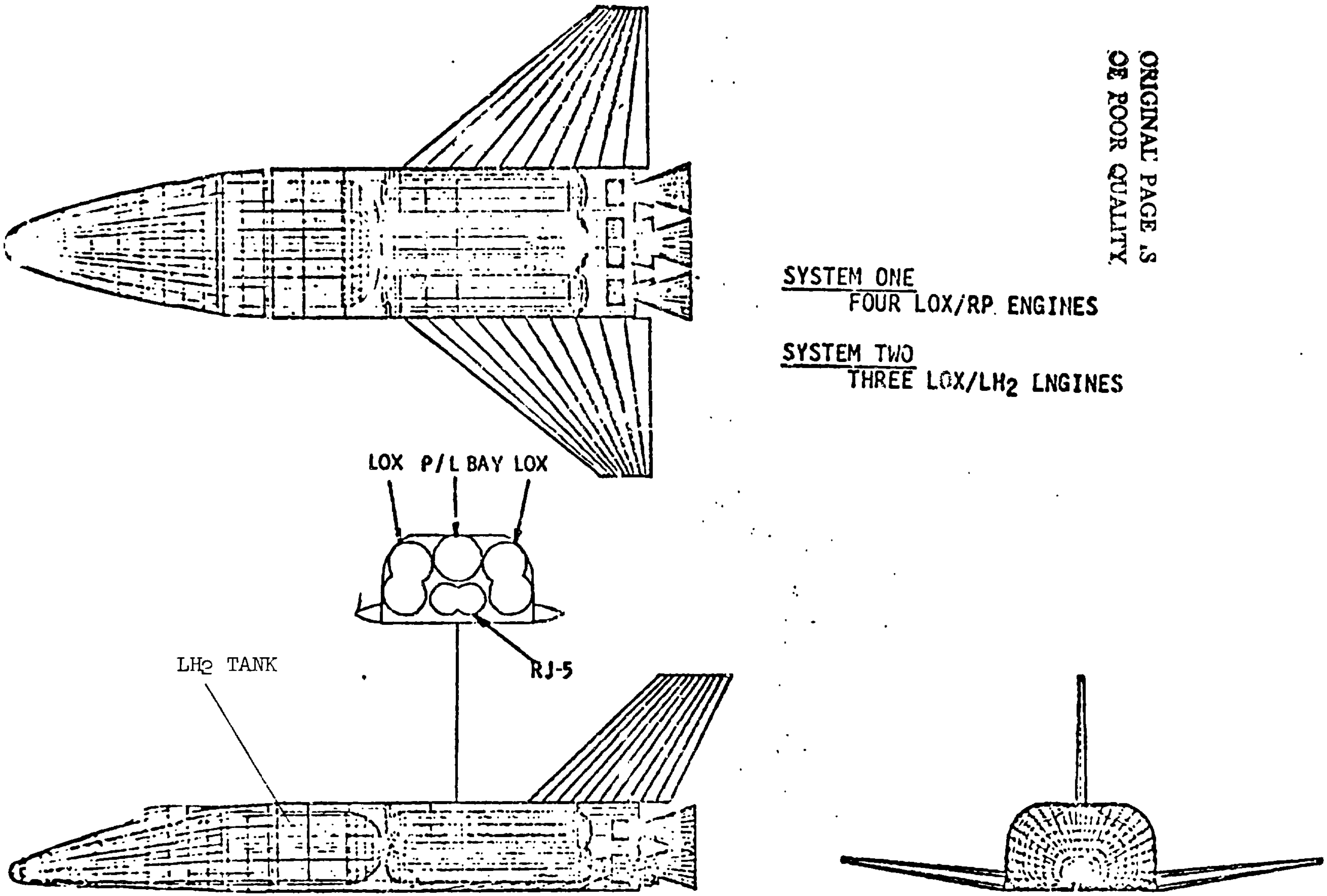


FIGURE 1.- SINGLE-STAGE-TO-ORBIT VEHICLE (EM 155). PORTIONS OF TOP SURFACE OF FOREBODY LH_2 TANK ARE EXPOSED. THE BOTTOM SURFACE IS PROTECTED WITH FAIRINGS AND METALLIC THERMAL PROTECTION. TWO TYPES OF PROPULSION SYSTEMS ARE INDICATED; THE LOX/RP SYSTEM BEING OPERATED IN PARALLEL WITH THE LOX/LH_2 SYSTEM EARLY IN THE FLIGHT. SUCH A PROPULSION SYSTEM GIVES AN OVERALL SMALLER VEHICLE EVEN THOUGH THE LOX/RP SYSTEM IS LOWER PERFORMING THAN LOX/LH_2.

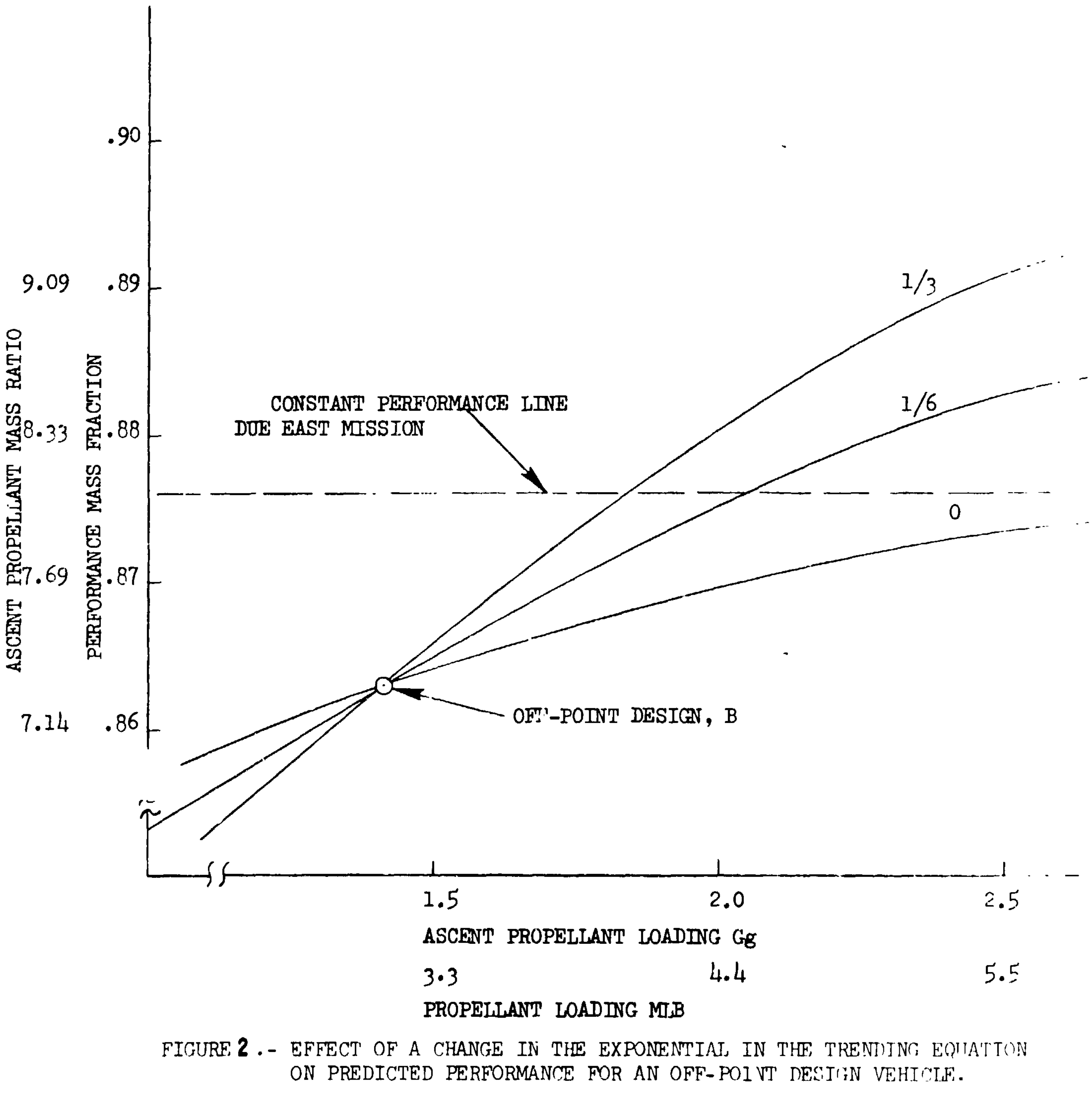


FIGURE 2 .- EFFECT OF A CHANGE IN THE EXPONENTIAL IN THE TRENDING EQUATION ON PREDICTED PERFORMANCE FOR AN OFF-POINT DESIGN VEHICLE.

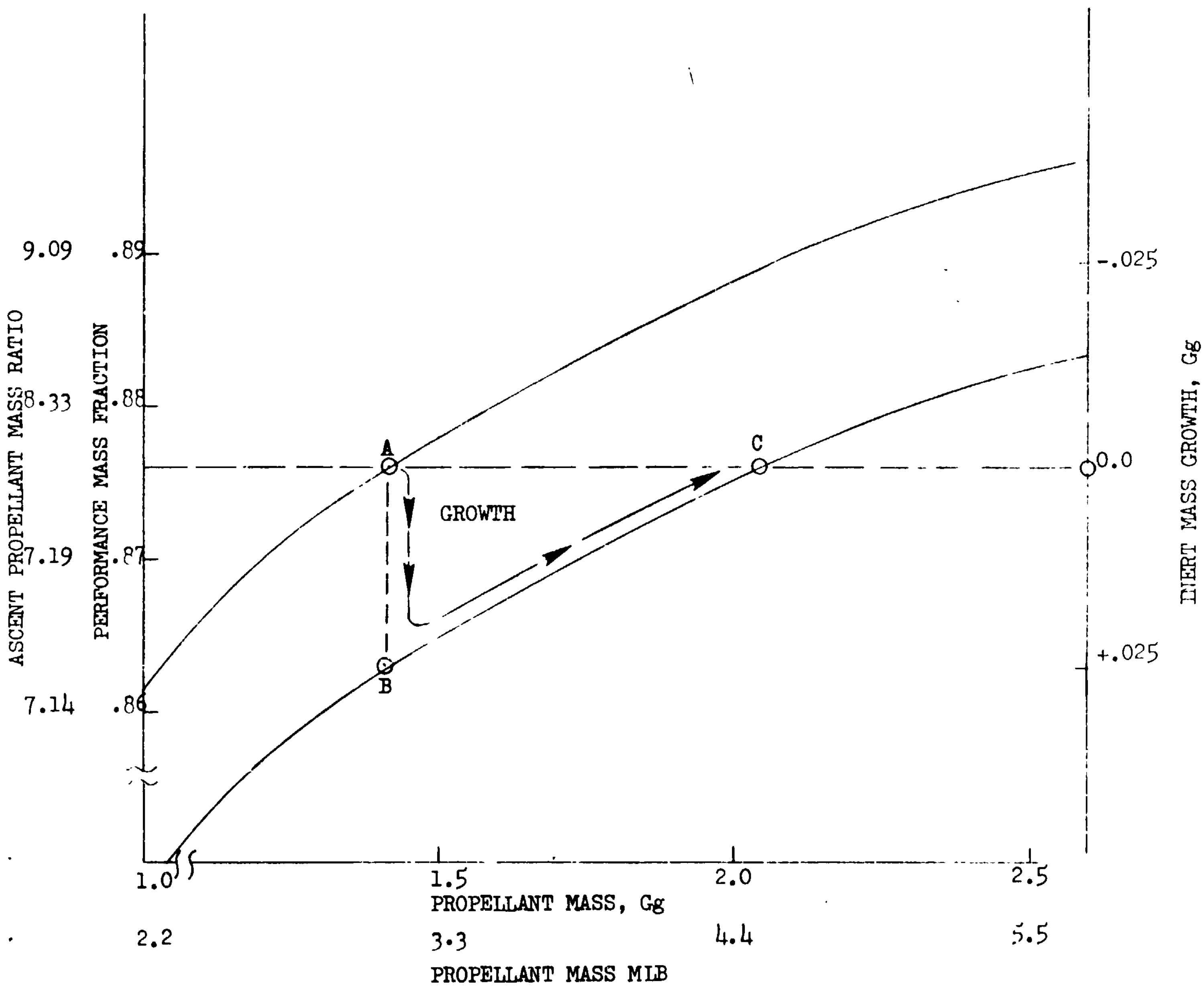


FIGURE 3.- VEHICLE TRENDING, EN 155 (29,500 Kg PAYLOAD)

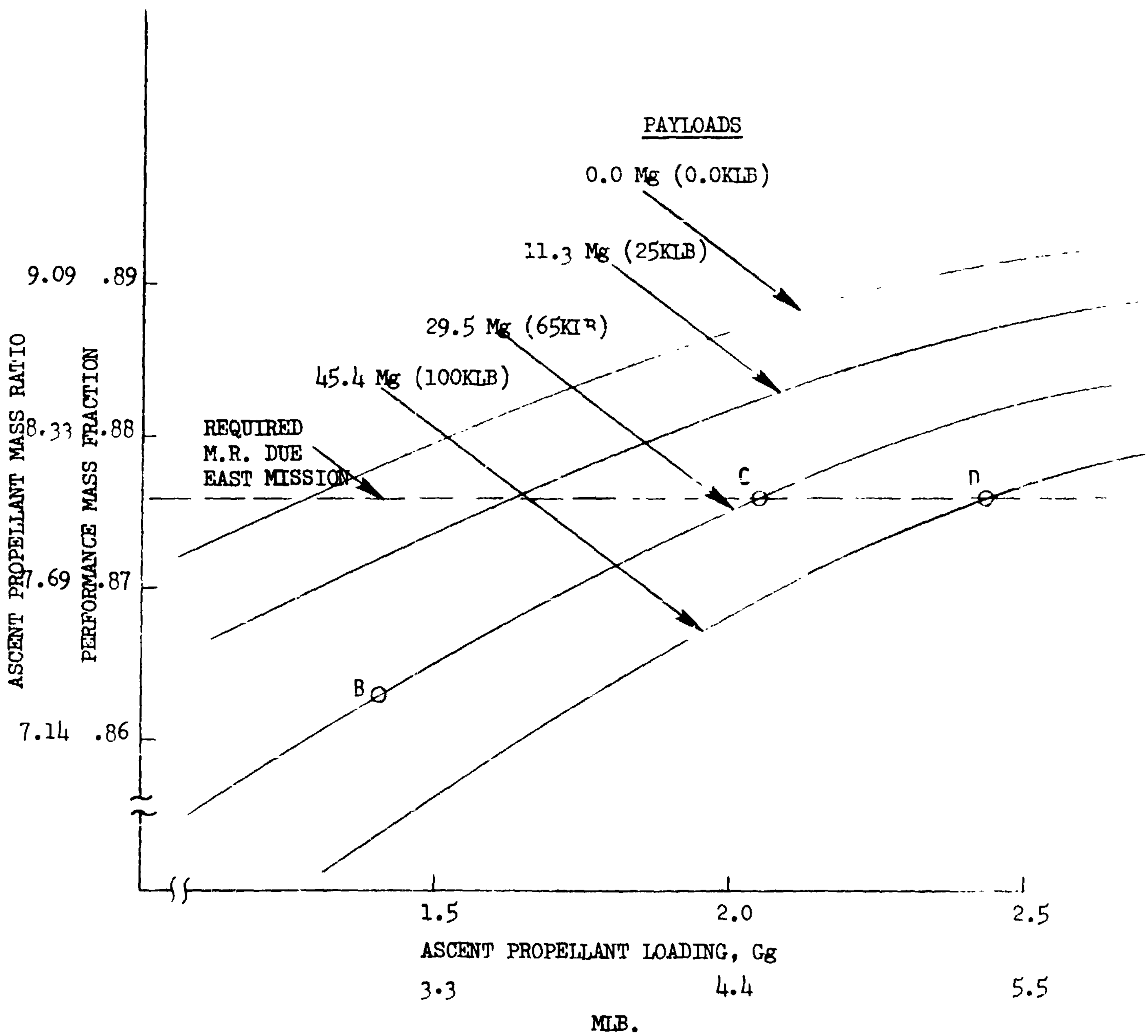


FIGURE 4.- PERFORMANCE MASS FRACTION VS. PROPELLANT LOADING FOR VARIOUS PAYLOADS.

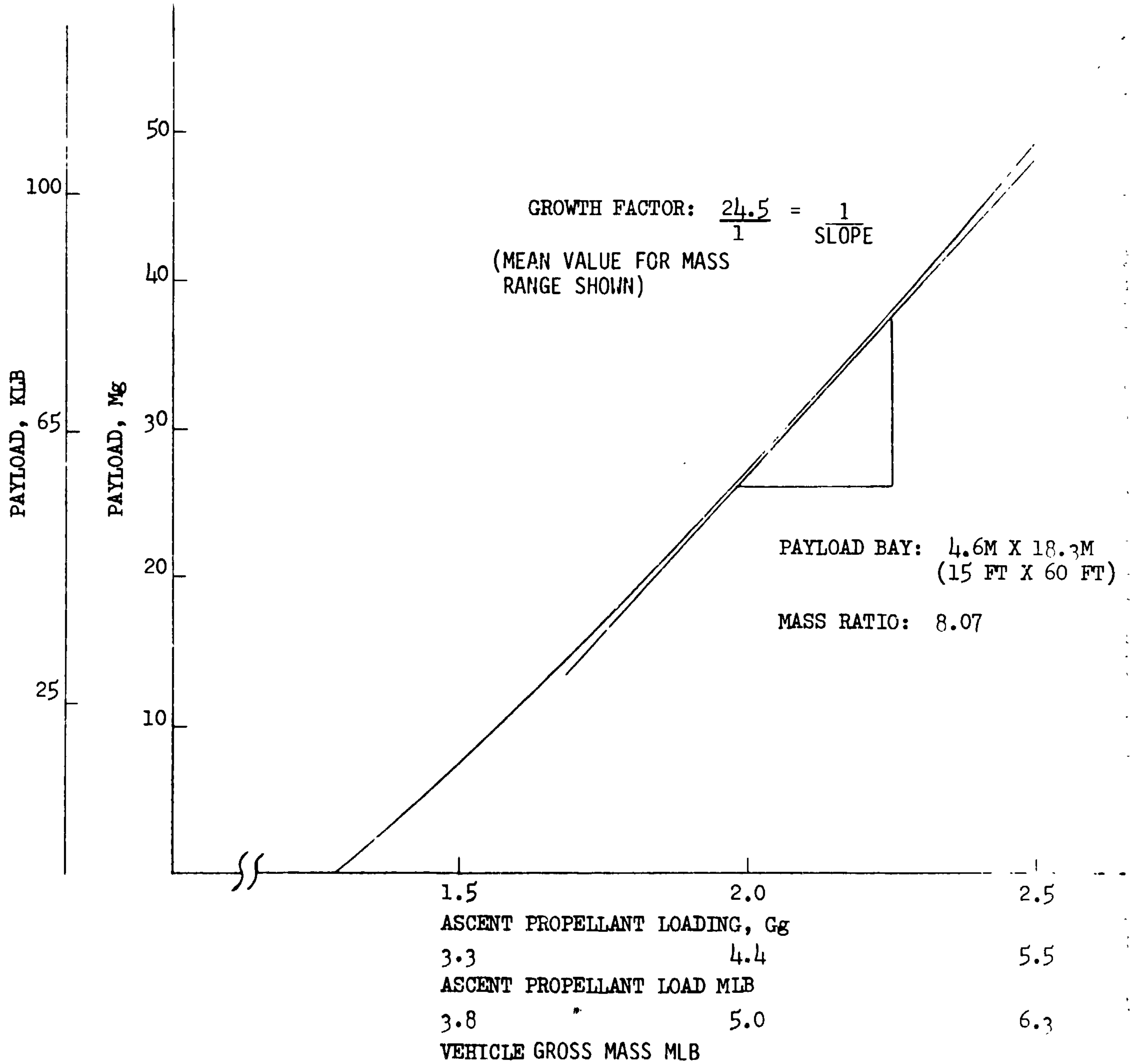


FIGURE 5.- EFFECT OF PAYLOAD MASS ON REQUIRED PROPELLANT LOADING.

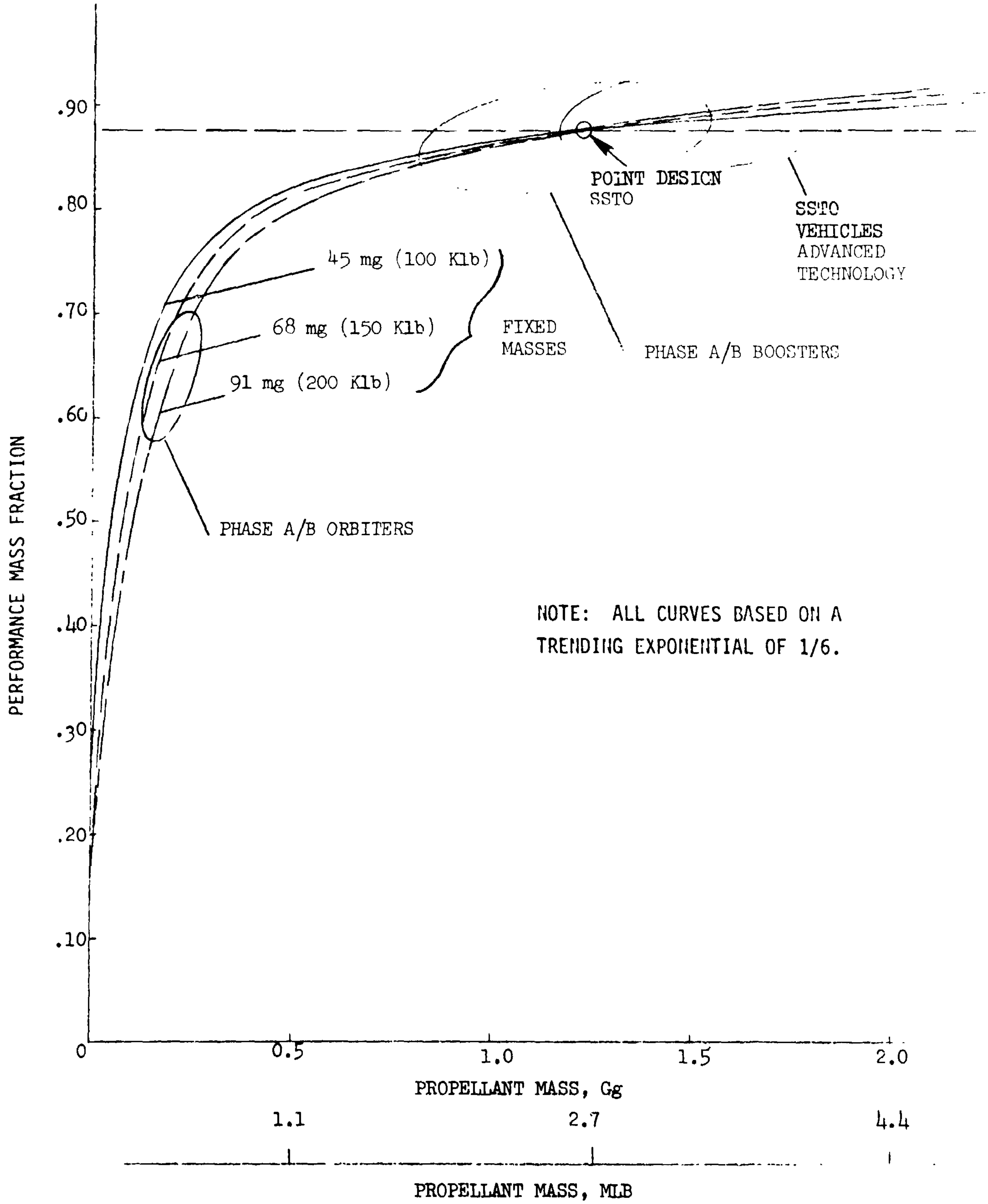


FIGURE 6 - EFFECT OF A CHANGE IN FIXED MASSES IN THE TRENDING EQUATION ON PREDICTED VEHICLE PERFORMANCE.

APPENDIX A

OVERALL VEHICLE TRENDING PROCEDURE AND SAMPLE CALCULATION

APPENDIX A

TRENDING PROCEDURE AND SAMPLE CALCULATION FOR A DUAL FUEL SSTO

1. Given: Gross Vehicle mass = 1,634,998 Kg

   Ascent propellant mass = 1,411,150 Kg

   Required m.r. = 8.1 λ = .8767

   Actual m.r. = 7.3 λ = .8631

2. Estimate Fixed Mass:

Payload =	29,500
Crew Compartment =	2,270
Avionics =	2,021
Manipulator =	349
Personnel and Provisions =	1500
Cargo Bay Doors and Cargo Bay Structure	9720
Total	45359 Kg

3. Compute **Trending Mass Fraction**

$$\lambda' = \frac{\text{ascent propellant}}{\text{gross - fixed mass}} = \frac{1,411,150}{1,634,998 - 45359}$$

$$= .8878$$

4. Compute vehicle point design characteristic, K_{V_1} based on point design propellant loading and **trending** mass fraction.

$$K_{V_1} = (m_{p'})^{1/6} \times \frac{1 - \lambda}{\lambda'}$$

$$= (1,411,150)^{1/6} \quad \frac{1 - .8878}{.8878}$$

$$= 1.3384$$

5. Compute λ for the new point design by assuming new ascent propellant loadings (or $m.r. = \frac{1}{1-\lambda}$)

or

$$\lambda = \frac{1}{1 + \frac{m_f}{m_{p_2}} + \left[\frac{1}{m_{p_2}}\right]^{1/6} K_{V_1}}$$

The required propellant mass is found to be 2,100,000 Kg compared to 1,411,150 Kg in the original vehicle.

or

$$\lambda = \frac{1}{1 + \frac{45,359}{2,100,000} + \left[\frac{1}{2,100,000}\right]^{1/6} 1.3384}$$

$$\lambda = \frac{1}{1.0216 + .1183} = .877$$

$$\text{and m.r.} = \frac{1}{1-.877} = 8.1$$

For changes in the required payload step 5 above is repeated. For example, assume the payload increment desired is 15,900 Kg. Then:

$$\lambda = \frac{1}{1 + \frac{45,359 + 15,900}{m_{p_2}} + \left[\frac{1}{m_{p_2}}\right]^{1/6} 1.3384}$$

The required λ is still assumed to be = .8767, or by iterative procedure on computer m_{p_2} is found to be equal to 2,437,500 Kg.

checking:

$$\lambda = \frac{1}{1 + \frac{61,259}{2,437,500} + \left[\frac{1}{2,437,500}\right]^{1/6} 1.3384}$$

$$\lambda = \frac{1}{1.0251 + .11537} = .877$$

TRENDING EQUATION PRINTOUT (EQ. 10)

```
K1= .453590E+05 K2= .133840E+01 WP1= .100000E+07 LAMBDA1= .848033E+00
K1= .453590E+05 K2= .133840E+01 WP1= .150000E+07 LAMBDA1= .865551E+00
K1= .453590E+05 K2= .133840E+01 WP1= .210000E+07 LAMBDA1= .877292E+00
K1= .453590E+05 K2= .133840E+01 WP1= .250000E+07 LAMBDA1= .882590E+00
K1= .453590E+05 K2= .133840E+01 WP1= .300000E+07 LAMBDA1= .887653E+00
K1= .612590E+05 K2= .133840E+01 WP1= .100000E+07 LAMBDA1= .836751E+00
K1= .612590E+05 K2= .133840E+01 WP1= .150000E+07 LAMBDA1= .857682E+00
K1= .612590E+05 K2= .133840E+01 WP1= .210000E+07 LAMBDA1= .871503E+00
K1= .612590E+05 K2= .133840E+01 WP1= .250000E+07 LAMBDA1= .877664E+00
K1= .612590E+05 K2= .133840E+01 WP1= .300000E+07 LAMBDA1= .883497E+00
```

K_1 corresponds to m_f in the text.

K_2 corresponds to Ky_1 in the text.

WP1 corresponds to m_{p_1} in the text.

ORIGINAL
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APPENDIX B

SUBSYSTEM TRENDING WITH EXAMPLES FOR THE SHUTTLE ORBITER AND LARC IN-HOUSE STUDY VEHICLE, EN-155

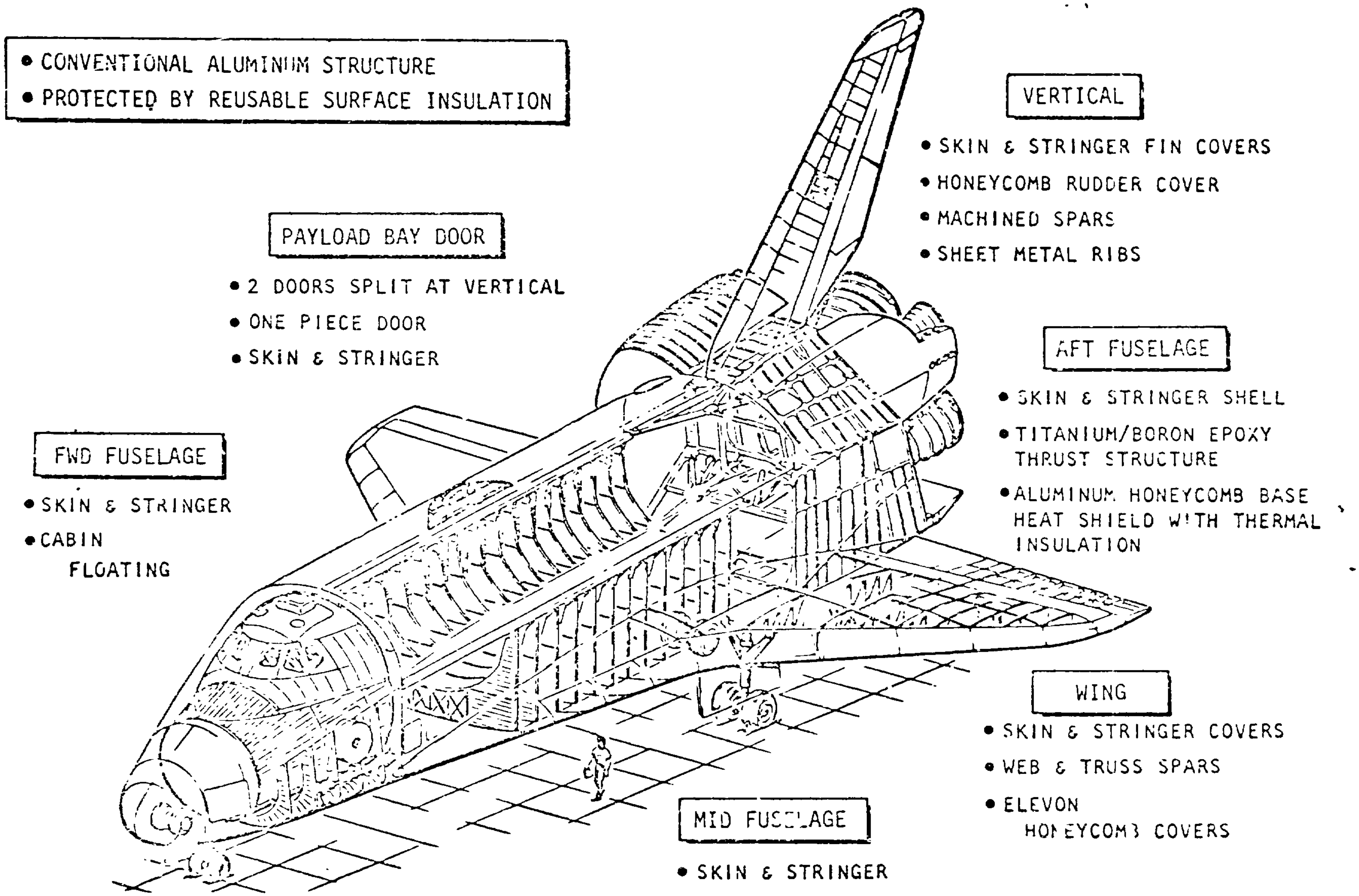

Figure B-1.- Vehicle test case for SEMP program.

REPRINT FROM ROCKWELL SPACE SHUTTLE SYSTEM SUMMARY DOCUMENT SEPTEMBER 1974.

SHUTTLE ORBITER
SUBSYSTEM TRENDING RESULTS

OK,

```
**************************
* M A S S   R E P O R T *
**************************
```

1.0 WING GROUP	15657.	LB	7102.	KG
2.0 TAIL GROUP	2911.	LB	1321.	KG
3.0 BODY GROUP	41961.	LB	19033.	KG
4.0 TPS	24380.	LB	11058.	KG
5.0 LANDING	8041.	LB	3647.	KG
6.0 PROPULSION	28200.	LB	12791.	KG
7.0 PROPULSION, RCS	2814.	LB	1276.	KG
8.0 PROPULSION, OMS	2976.	LB	1350.	KG
9.0 PRIME POWER	3030.	LB	1374.	KG
10.0 ELEC CONV AND DISTR	7310.	LB	3316.	KG
11.0 HYDRAULICS	1775.	LB	805.	KG
12.0 SURFACE CONTROLS	2615.	LB	1186.	KG
13.0 AVIONICS	6011.	LB	2727.	KG
14.0 ENVIRONMENTAL CONTROL	5270.	LB	2390.	KG
15.0 PERSONNEL PROVISIONS	1021.	LB	463.	KG
16.0 MARGIN	767.	LB	348.	KG
DRY WEIGHT	154739.	LB	70188.	KG
17.0 PERSONNEL	2640.	LB	1197.	KG
18.0 PAYLOAD ACCOM.	1608.	LB	729.	KG
19.0 CARGO (RETURNED)	32000.	LB	14515.	KG
20.0 RESIDUAL FLUIDS	1551.	LB	704.	KG
LANDED WEIGHT	192538.	LB	87334.	KG
21.0 OMS AND RCS RESERVES	77.	LB	35.	KG
OMS	58	LB	26.	KG
RCS	19.	LB	9.	KG
ENTRY WEIGHT	192615.	LB	87369.	KG
22.0 RCS PROPELLANT (ENTRY)	828.	LB	376.	KG
DESCENT WEIGHT	193444.	LB	87745.	KG
23.0 ACPS CONSUMABLES (RCS + OMS) ON ORBIT	14547.	LB	6598.	KG
RCS	1664.	LB	755.	KG
OMS	12883.	LB	5844.	KG
24.0 CARGO DISCHARGED	23000.	LB	14969.	KG
INJECTED WEIGHT	240990.	LB	109311.	KG
25.0 ASCENT RESERVES AND ASCENT PROPELLANT RESIDUALS	4454.	LB	2020.	KG
26.0 INFLIGHT LOSSES	662.	LB	300.	KG
27.0 ASCENT PROPELLANT	5206.	LB	2361.	KG
GROSS LIFT OFF WEIGHT	261313.	LB	113994.	KG

SHUTTLE DATA

SUBSYSTEM TRENDING RESULTS

OK,

```
****************************
* M A S S   R E P O R T *
****************************
```

1.0 WING GROUP	67330.	LB	30567.	KG
2.0 TAIL GROUP	8438.	LB	3827.	KG
3.0 BODY GROUP	161073.	LB	73062.	KG
4.0 TPS	6369.	LB	2889.	KG
5.0 LANDING	15839.	LB	7184.	KG
6.0 PROPULSION	95710.	LB	43413.	KG
7.0 PROPULSION, RCS	13737.	LB	6231.	KG
8.0 PROPULSION, OMS	5299.	LB	2403.	KG
9.0 PRIME POWER	2487.	LB	1128.	KG
10.0 ELEC CONV AND DIST	10023.	LB	4546.	KG
11.0 HYDRAULICS	1929.	LB	875.	KG
12.0 SURFACE CONTROLS	2962.	LB	1344.	KG
13.0 AVIONICS	3596.	LB	1631.	KG
14.0 ENVIRONMENTAL CON	6907.	LB	3133.	KG
15.0 PERSONNEL PROVIS	1021.	LB	463.	KG
16.0 MARGIN	30707.	LB	13928.	KG
DRY WEIGHT	433484.	LB	196625.	KG
17.0 PERSONNEL	2640.	LB	1197.	KG
18.0 PAYLOAD ACCOM.	100.	LB	45.	KG
19.0 CARGO (RETURNED)	65020.	LB	29454.	KG
20.0 RESIDUAL FLUIDS	63.	LB	29.	KG
LANDED WEIGHT	[illegible]	LB	227380.	KG
21.0 OMS AND RCS RESERVES	[illegible]	LB	68.	KG
OMS	100.	LB	45.	KG
RCS	50.	LB	23.	KG
ENTRY WEIGHT	501437.	LB	227448.	KG
22.0 RCS PROPELLANT (ENTRY)	2607.	LB	1183.	KG
DESCENT WEIGHT	504245.	LB	228631.	KG
23.0 ACPS CONSUMABLES (RCS + OMS) ON ORBIT	26916.	LB	12263.	KG
RCS	3276.	LB	1486.	KG
OMS	23640.	LB	10723.	KG
24.0 CARGO DISCHARGED	0.	LB	0.	KG
INJECTED WEIGHT	530961.	LB	240840.	KG
25.0 ASCENT RESERVES AND ASCENT PROPELLANT RESIDUALS	17653.	LB	8007.	KG
26.0 INFLIGHT LOSSES	13326.	LB	6045.	KG
27.0 ASCENT PROPELLANT	3099171.	LB	1405760.	KG
GROSS LIFT OFF WEIGHT	3661111.	LB	1660652.	KG

VTOHL SSTO DUAL-FUELED VEHICLE (EN-155) LREF=201.5 FT.

APPENDIX C

SUBSYSTEM TRENDING SAMPLE COMPUTER INPUT DATA

AND COMPUTER PROGRAM

COMPUTER INPUT DATA
(SAMPLE)

```
$LIST WERDAT
GO
 $WERDAT
 N=2,
 RPH(1)=5.18,5.18,
 RH(1)=.00727,.015,
 EPS1(1)=40.,40.,
 RNE(1)=6.054,9.543,
 EPS2(1)=0.,200.,
 PC(1)=3000.,3000.,
 RNA(1)=36.85,60.54,
 AMDOT(1)=2131.,1732.4,
 RPF(1)=.59,.59,
 RGA(1)=.00129,.00129,
 AISE(1)=345.,466.,
 AME(1)=3.98,3.05,
 $END
3.75        XNZ
.2          FW
5119.       SBPLAN
4678.       SWEXP
7.63        TROOT
.453        XKW
101.65        BESU
.1          CW
47.748        BWIDTH
1.108       XKT
1350.         ST
1740.       AB
4.          XNCREW
3.4         BD
14086.       SBWET
.766          CB
18355.          V1
.843          DB
60387.          V2
.7          EB
4819.          V3
.0024       FB
1.59        GB
478.        HBF
.01         XKTPS
.085          TBAR
12798.       STPLAN
.529         CL
35155.      STWET
.030        XKLAND
805.        WM
0.        WS
.0125         E1
735300.     TUAC1
3.98          XENG1
.0195          E2
807328.          TVAC2
3.05         XENG2
.000136     XKRCS
201.5       XLREF
.0350       AOMS

26283.          TVOMS
.152            BOMS
22762.          WPOMS
0.              COMS
.610            APRIME
832.            AC
.000097         BPRIME
.425            CPRIME
.020            XKECD
1.23             AHCD
.000168           BHCD
3.32            ASC
202.            BSC
713.          WA.
8060.           VEC
5.              DAV
353.            AFEHF
167.            BFERF
.1              XKMAR
400.            EFER
560.            FFER
100.          WFAYPR
65000.         WPAYLD
.0009             XKRJF
.0002           XKOMSR
.0001           XKRCSR
.0052           XKRCSE
.0469           XKOMSC
.0065           XKRCSC
0.          WPAYDE
.0099           ARR
.0040           BRR
.0043         XKIFL
3099171.        WAPROP
1.              AICDE
2.0             XNEC

CK,
```

```
*100
     4(WPERP,WT(15)),
     5(WMAR,WT(16)),
     6(WDRY,WT(17)),
     7(WFER,WT(18)),
     8(WPYFR2,WT(19)),
     9(WPYLD2,WT(20))
      EQUIVALENCE (WRUF,WT(21)),
     1(WLAND,WT(22)),
     2(WORTOT,WT(23)),
     3(WOMSR,WT(34)),
     4(WRCSR,WT(35)),
     5(WENTR2,WT(24)),
     6(WRCSE,WT(25)),
     7(WDESC,WT(26)),
     8(WOMRCT,WT(27)),
     9(WOMSC,WT(37))
      EQUIVALENCE (WRCSC,WT(36)),
     1(WPYDE2,WT(28)),
     2(WINJ,WT(29)),
     3(WRR,WT(30)),
     4(WIFL,WT(31)),
     5(WPROP2,WT(32)),
     6(WGROSS,WT(33))
      EQUIVALENCE (WWINGM,WT(38)),
     1(WTAILM,WT(39)),
     2(WBODYM,WT(40)),
     3(WTPSM,WT(41)),
     4(WLDRM,WT(42)),
     5(WPROPM,WT(43)),
     6(WRCSM,WT(44)),
     7(WOMSM,WT(45)),
     8(WFRIMM,WT(46)),
     9(WECDM,WT(47))
      EQUIVALENCE (WHCDM,WT(48)),
     1(WSCM,WT(49)),
     2(WAU2M,WT(50)),
     3(WENUM,WT(51)),
     4(WPERPM,WT(52)),
     5(WMARM,WT(53)),
     6(WDRYM,WT(54)),
     7(WPERM,WT(55)),
     8(WPYPRM,WT(56)),
     9(WPYLDM,WT(57))
      EQUIVALENCE (WRUFM,WT(58)),
     1(WLANDM,WT(59)),
     2(WORTM,WT(60)),
     3(WOMSRM,WT(61)),
     4(WRCSRM,WT(62)),
     5(WENTRM,WT(63)),
     6(WRCSEM,WT(64)),
     7(WDESCM,WT(65)),
     8(WOMRCM,WT(66)),
     9(WOMSCM,WT(67))
      EQUIVALENCE (WRCSCM,WT(68)),
     1(WPYDEM,WT(69)),
     2(WINJM,WT(70)),
     3(WRRM,WT(71)),
     4(WIFLM,WT(72)),
     5(WPRF2M,WT(73)),
     6 WOPOSM,WT(74))
      DATA WERDAT/
     1'WERDAT',
     2'N       ',2,0,1,1,'RPH     ',3,0,1,2,
     3'RN      ',3,0,1,2,'EPS1    ',3,0,1,2,
     4'RNE     ',3,0,1,2,'EPS2    ',3,0,1,2,
     5'PC      ',3,0,1,2,'RNA     ',3,0,1,2,
     6'AMDOT   ',3,0,1,2,'RPF     ',3,0,1,2,
     7'RGA     ',3,0,1,2,'AISE    ',3,0,1,2,
     8'ANE     ',3,0,1,2,0/
      WERDAT( 8) = LOC(N)
      WERDAT(15) = LOC(RPH)
      WERDAT(22) = LOC(RN)
      WERDAT(29) = LOC(EPS1)
      WERDAT(36) = LOC(RNE)
      WERDAT(43) = LOC(EPS2)
      WERDAT(50) = LOC(PC)
      WERDAT(57) = LOC(RNA)
      WERDAT(64) = LOC(AMDOT)
      WERDAT(71) = LOC(RPF)
      WERDAT(78) = LOC(RGA)
      WERDAT(85) = LOC(AISE)
      WERDAT(92) = LOC(ANE)
C     NAMELIST/NAM1/N,RPH,RN,EPS1,RNE,EPS2,PC,RNA,AMDOT,RPF,R
CGA,AISE,ANE
C
C           INPUT VARIABLES
C
C
C     TRACE 888
C     IF AICODE=1.0 , WILL CALCULATE WPROP USING NEW EQUATION

C     WRITE(1,953)
C     READ(10,800) TITLE
C     READ(10,NAM1)
C     WRITE(1,NAM1)
      CALL NAMIN(5,WERDAT)
      DO 10 I=1,77
      READ(5,900) VAR(I),VARN(I)
      VAR(I) = VAR(I)
C     WRITE(1,952) VAR(I),VARN(I)
   10 CONTINUE
C 800 FORMAT(8A10)
  900 FORMAT(F15.0,4X,5A2)
```

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```
100
  900 FORMAT(F15.0,4X,5A2)
C 952 FORMAT(F15.4,4X,A10)
C 953 FORMAT(1H1,14H**INPUT DATA**,//)
C
      WLAND=450000.
      WENTRY=450000.
      WDRY=400000.
    1 CONTINUE
C
C        WING EQU
      WWING= (XNZ*WLAND/(1. + FU*SPLAN/SWEXP))**.386*
     1        (SWEXP/TROOT)**.572*(XKW*BESW**.572 + CU*BWIDTH**.572)
      WWINGM=WWING*0.4535924
C        TAIL EQU
      WTAIL = XKT*ST**1.24
      WTAILM=WTAIL*0.4535924
C        BODY
      TVAC = TVAC1*XENG1 + TVAC2*XENG2
      WBODY = AB*XNCREW**.5 + BB*SBWET*XNZ**.3333 +
     1        CB*V1 + DB*V2 + EB*V3 + FB*TVAC + GB*ABF**1.15
      WBODYM=WBODY*0.4535924
      X = AB*XNCREW**.5
      X = BB*SBWET*XNZ**.33333
      X = CB*V1
      X = DB*V2
      X = EB*V3
      X = FB*TVAC
      X = GB*ABF**1.15
C        TPS
      WTPS=XKTPS/TBAR**.302*(WENTRY/STPLAN/CL)**.5*STWET
      WTPSM=WTPS*0.4535924
C        LAND, DOCK, RECOVER
      WLDR = XKLAND*WLAND + WM + WS
      WLDRM=WLDR*0.4535924
C        ASCENT PROPULSION
      IF(AICODE .EQ. 1.) GO TO 20
      WPROP = E1*TVAC1*XENG1 + E2*TVAC2*XENG2
      GO TO 30
   20 WPROP=0.0
      DO 40 II=1,N
      APART=EPS1(II)-1.0
      BPART=EPS2(II)-EPS1(II)
      CPART=SQRT(EPS2(II))-1.
      DPART=SQRT(AMDOT(II)*PC(II))
      WPRP(II)=(RPH(II)+RN(II)*APART+RNE(II)*BPART/PC(II)+RNA(II)
     1*CPART/DPART+RPS(II)+RCA(II)*AISE(II))*ANE(II)*AMDOT(II)
      WPROP=WPROP+WPRP(II)
      WRITE(1,801) II,WPROP
   40 CONTINUE
  801 FORMAT(///,10X,'PROPULSION',I2, '=',F15.4)
   30 WPROPM=WPROP*0.4535924
C        PROP RCS
      WRCS = XKRCS*WENTRY*XLREF
      WRCSM=WRCS*0.4535924
C        PROP OMS
      WOMS=AOMS*XNEO*TVOMS+BOMS*WPOMS
      WOMSM=WOMS*0.4535924
C        ELECT CON AND DIS
      WECD=XKECD*WLAND
      WECDM=WECD*0.4535924
C        HYDRAULIC CON AND DIS
      WHCD = AHCD*AC + BHCD*TVAC
      WHCDM=WHCD*0.4535924
C        SURFACE CONTROLS
      WSC = ASC*AC + BSC
      WSCM=WSC*0.4535924
C        AVIONICS
      WAV2 = WAV * WDRY**.125
      WAV2M=WAV2*0.4535924
C        PRIME POWER
      WPRIME = APRIME*AC + BPRIME*TVAC + CPRIME*WAV2
      WPRIMM=WPRIME*0.4535924
C        ENVIRONMENTAL CONTROL
      WENV=5.85*VEC**.75 + 10.5*XNCREW*DAY + 0.44*WAV2
      WENVM=WENV*0.4535924
C        PERSONAL PROVISIONS
      WPERP = APERP + BPERP*XNCREW
      WPERPM=WPERP*0.4535924
C        WDRY
      WDRY = WWING + WTAIL + WBODY + WTPS + WLDR + WPROP + WR
     CS +
     1        WOMS + WPRIME + WECD + WHCD + WSC + WAV2+ WENV +
     2        WPERP
      WDRYM=WDRY*0.4535924
C        MARGIN
      WMAR=XKMAR*(WDRY-WPROP)
      WMARM=WMAR*0.4535924
C        ADD TO WDRY
      WDRY = WDRY + WMAR
      WDRYM=WDRY*0.4535924
C        PERSONNEL
      WPER = EPER + FPER*XNCREW
      WPERM=WPER*0.4535924
C        PAYLOAD PROVISIONS
      WPYPR2 = WPAYPR
      WPYPRM=WPYPR2*0.4535924
C        PAYLOAD
      WPYLD2 = WPAYLD
      WPYLDM=WPYLD2*0.4535924
C        RESIDUALS AND UNUSABLE FLUIDS
      WRUF=XKRUF * WENTRY**.85
```

```
P100
      WRUF=XKRUF * WENTRY**.85
      WRUFM=WRUF*0.4535924
C       ADD TO LANDED WEIGHT
      WLAND = WDRY + WPER + WPAYPR + WPAYLD + WRUF
      WLANDM=WLAND*0.4535924
C         RESERVES OMS AND RCS
      WOMSR = WLAND*XKOMSR
      WOMSRM=WOMSR*0.4535924
      WRCSR = WLAND*XKRCSR
      WRCSRM=WRCSR*0.4535924
C        ENTRY WEIGHT
      WENTR2 = WLAND + WOMSR + WRCSR
      WENTRM=WENTR2*0.4535924
C         RCS PROP ENTRY
      WRCSE = WENTR2*XKRCSE
      WRCSEM=WRCSE*0.4535924
C         DESCENT WEIGHT
      WDESC = WENTR2 + WRCSE
      WDESCM=WDESC*0.4535924
C        RCS AND OMS PROP CONSUMABLES
      WOMSC = WDESC*XKOMSC
      WOMSCM=WOMSC*0.4535924
      WRCSC = WDESC*XKRCSC
      WRCSCM=WRCSC*0.4535924
C         DELIVERED CARGO
      WPYDE2 = WPAYDE
      WPYDEM=WPYDE2*0.4535924
C         INJECTED WEIGHT
      WINJ = WDESC + WOMSC + WRCSC + WPAYDE
      WINJM=WINJ*0.4535924
C         ASCENT RESERVES AND ASCENT RESIDUALS
      WRR = WINJ*ARR + BRR*WAPROP
      WRRM=WRR*0.4535924
C         INFLIGHT LOSES
      WIFL = XKIFL*WAPROP
      WIFLM=WIFL*0.4535924
C         ASCENT PROPELLANT
      WPROP2=WAPROP
      WPPP2M=WPROP2*0.4535924
C         GROSS LIFTOFF WEIGHT
      WGROSS = WINJ + WRR + WIFL + WPROP2
      WGROSM=WGROSS*0.4535924
C         WENTRY LOOP
      IF (ABS(WENTRY-WENTR2)/WENTR2 .LT. .001)  GO TO 999
      WENTRY = WENTR2
  988 CONTINUE
      GO TO 1
  999 CONTINUE
      CALL ERASE
      CALL WPRNT
      CALL EXIT
      END
      SUBROUTINE WPRNT
      COMMON/UTS/UT(100),TITLE(40)
C     WRITE(1,954)
C 954 FORMAT(1H1)
C     WRITE(1,970) TITLE
C 970 FORMAT(40A2)
      WRITE (1,950)
  950 FORMAT(//)
      WRITE(1,940)
  940 FORMAT(53X,25H*************************)
      WRITE (1,941)
  941 FORMAT(53X,25H* M A S S   R E P O R T *)
      WRITE(1,940)
      WRITE(1,942)
  942 FORMAT(//)
      WRITE(1,900) UT(1), UT(38)
  900 FORMAT(16X,' 1.0 WING GROUP',39X,F15.0,2X,'LB',5X,F15.0
     ,2X,'KG')
      WRITE(1,901) UT(2), UT(39)
  901 FORMAT(16X,' 2.0 TAIL GROUP',39X,F15.0,2X,'LB',5X,F15.0
     ,2X,'KG')
      WRITE(1,902) UT(3) ,UT(40)
  902 FORMAT(16X,' 3.0 BODY GROUP',39X,F15.0,2X,'LB',5X,F15.0
     ,2X,'KG')
      WRITE(1,903) UT(4), UT(41)
  903 FORMAT(16X,' 4.0 TPS',46X,F15.0,2X,'LB',5X,F15.0,2X,'KG
     ')
      WRITE(1,904) UT(5),UT(42)
  904 FORMAT(16X,' 5.0 LANDING',42X,F15.0,2X,'LB',5X,F15.0,2X
     ,'KG')
      WRITE(1,905) UT(6),UT(43)
  905 FORMAT(16X,' 6.0 PROPULSION',39X,F15.0,2X,'LB',5X,F15.0
     ,2X,'KG')
      WRITE(1,906) UT(7),UT(44)
  906 FORMAT(16X,' 7.0 PROPULSION, RCS',34X,F15.0,2X,'LB',5X,
     F15.0,2X,
     1'KG')
      WRITE(1,907) UT(8),UT(45)
  907 FORMAT(16X,' 8.0 PROPULSION, OMS',34X,F15.  2X,'LB',5X,
     F15.0,2X,
     1'KG')
      WRITE(1,908) UT(9),UT(46)
  908 FORMAT(16X,' 9.0 PRIME POWER',38X,F15.0,2X,'LB',5X,F15.
     0,2X,'KG')
      WRITE(1,909) UT(10), UT(47)
  909 FORMAT(16X,'10.0 ELEC CONV AND DISTR',30X,F15.0,2X,'LB'
     ,5X,F15.0,
     12X,'KG')
      WRITE(1,910) UT(11),UT(48)
  910 FORMAT(16X,'11.0 HYDRAULICS',39X,F15.0,2X,'LB',5X,F15.0
     ,2X,'KG')
      WRITE(1,911) UT(12),UT(49)
  911 FORMAT(16X,'12.0 SURFACE CONTROLS',33X,F15.0,2X,'LB',5X
     ,F15.0,2X,
     1'KG')
      WRITE(1,912) UT(13), UT(50)
  912 FORMAT(16X,'13.0 AVIONICS',41X,F15.0,2X,'LB',5X,F15.0,2
     X,'KG')
      WRITE(1,913) UT(14),UT(51)
  913 FORMAT(16X,'14.0 ENVIRONMENTAL CONTROL',28X,F15.0,2X,'L
     B',5X,
     1F15.0,2X,'KG')
```

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```
P50
     1F15.0,2X,'KG')
      WRITE(1,914) WT(15), WT(52)
  914 FORMAT(16X,'15.0 PERSONNEL PROVISIONS',29X,F15.0,2X,'LB',5X,F15.0,
     12X,'KG')
      WRITE(1,915) WT(16),WT(53)
  915 FORMAT(16X,'16.0 MARGIN',43X,F15.0,2X,'LB',5X,F15.0,2X,'KG',//)
      WRITE(1,916) WT(17),WT(54)
  916 FORMAT(26X,'DRY WEIGHT',34X,F15.0,2X,'LB',5X,F15.0,2X,'KG',//)
      WRITE(1,917) WT(18),WT(55)
  917 FORMAT(16X,'17.0 PERSONNEL',40X,F15.0,2X,'LB',5X,F15.0,2X,'KG')
      WRITE(1,918) WT(19),WT(56)
  918 FORMAT(16X,'18.0 PAYLOAD ACCOM.',35X,F15.0,2X,'LB',5X,F15.0,2X,
     1'KG')
      WRITE(1,919) WT(20),WT(57)
  919 FORMAT(16X,'19.0 CARGO (RETURNED)',33X,F15.0,2X,'LB',5X,F15.0,2X,
     1'KG')
      WRITE(1,920) WT(21),WT(58)
  920 FORMAT(16X,'20.0 RESIDUAL FLUIDS',34X,F15.0,2X,'LB',5X,F15.0,2X,
     1'KG',//)
      WRITE(1,921) WT(22),WT(59)
  921 FORMAT(26X,'LANDED WEIGHT',31X,F15.0,2X,'LB',5X,F15.0,2X,'KG',//)
      WORTOT=WT(34)+WT(35)
      WORTM=WORTOT*0.4535924
      WRITE(1,922) WORTOT,WORTM
  922 FORMAT(16X,'21.0 OMS AND RCS RESERVES',29X,F15.0,2X,'LB',5X,F15.0,
     12X,'KG')
      WRITE(1,933) WT(34),WT(61)
  933 FORMAT(32X,'OMS',35X,F15.0,2X,'LB',5X,F15.0,2X,'KG')
      WRITE(1,934) WT(35),WT(62)
  934 FORMAT(32X,'RCS',35X,F15.0,2X,'LB',5X,F15.0,2X,'KG',//)
      WRITE(1,923) WT(24),WT(63)
  923 FORMAT(26X,'ENTRY WEIGHT',32X,F15.0,2X,'LB',5X,F15.0,2X,'KG',//)
      WRITE(1,924) WT(25),WT(64)
  924 FORMAT(16X,'22.0 RCS PROPELLANT (ENTRY)',27X,F15.0,2X,'LB',5X,
     1F15.0,2X,'KG',//)
      WRITE(1,925) WT(26),WT(65)
  925 FORMAT(26X,'DESCENT WEIGHT',30X,F15.0,2X,'LB',5X,F15.0,2X,'KG',//)
      WOMRT=WT(36)+WT(37)
      WOMRTM=WOMRT*0.4535924
      WRITE(1,926) WOMRT,WOMRTM
  926 FORMAT(16X,'23.0 ACPS CONSUMABLES (RCS + OMS) ON ORBIT',12X,F15.0,
     12X,'LB',5X,F15.0,2X,'KG')
      WRITE(1,935) WT(36),WT(68)
  935 FORMAT(32X,'RCS',35X,F15.0,2X,'LB',5X,F15.0,2X,'KG')
      WRITE(1,936) WT(37), WT(67)
  936 FORMAT(32X,'OMS',35X,F15.0,2X,'LB',5X,F15.0,2X,'KG')
      WRITE(1,927) WT(28), WT(69)
  927 FORMAT(16X,'24.0 CARGO DISCHARGED',33X,F15.0,2X,'LB',5X,F15.0,2X,
     1'KG',//)
      WRITE(1,928) WT(29),WT(70)
```

```
350
      WRITE(1,928) WT(29),WT(70)
  928 FORMAT(26X,'INJECTED WEIGHT',29X,F15.0,2X,'LB',5X,F15.0,2X,'KG'//)
      WRITE(1,929) WT(30), WT(71)
  929 FORMAT(16X,'25.0 ASCENT RESERVES AND ASCENT PROPELLANT RESIDUALS',
     12X,F15.0,2X,'LB',5X,F15.0,2X,'KG')
      WRITE(1,930) WT(31),WT(72)
  930 FORMAT(16X,'26.0 INFLIGHT LOSSES',34X,F15.0,2X,'LB',5X,F15.0,2X,
     1'KG')
      WRITE(1,931) WT(32),WT(73)
  931 FORMAT(16X,'27.0 ASCENT PROPELLANT',32X,F15.0,2X,'LB',5X,F15.0,2X,
     1'KG',//)
      WRITE(1,932) WT(33),WT(74)
  932 FORMAT(26X,'GROSS LIFT OFF WEIGHT',23X,F15.0,2X,'LB',5X,F15.0,2X,
     1'KG')
      WRITE(1,888)
  888 FORMAT(//,30X,'VTOHL SSTO DUAL-FUELED VEHICLE (EN-155) LREF=201.5
     1FT.')
      RETURN
      END
BOTTOM
```

SEMP PRINTOUT

(SYSTEMS ENGINEERING MASS PROPERTIES COMPUTER PROGRAM)

```
ED KWER
GO
EDIT
V
P100
.NULL.
C     PROGRAM WER(INPUT,OUTPUT,TAPE10=INPUT,TAPE1=OUTPUT)
      COMMON / INT / VAR(100)
      COMMON/UTS/UT(100),TITLE(40)
      DIMENSION VARN(100)
      DIMENSION RPH(2),RN(2),EPS1(2),RNE(2),EPS2(2),PC(2),RNA(2),
     1AMDOT(2),RPF(2),RGA(2),AISE(2),ANE(2),WPRP(2)
      INTEGER WERDAT(55)
      EQUIVALENCE (VAR( 1),XNZ),
     1 (VAR( 2),FW),
     1 (VAR( 3),SBPLAN),
     1 (VAR( 4),SWEXP),
     1 (VAR( 5),TROOT),
     1 (VAR( 6),XKW),
     1 (VAR( 7),BESW),
     1 (VAR( 8), CW),
     1 (VAR( 9),BWIDTH),
     1 (VAR(10),XKT),
     1 (VAR(11),ST)
      EQUIVALENCE (VAR(12),AB),
     1 (VAR(13),XNCREW),
     1 (VAR(14),BB),
     1 (VAR(15),SBWET),
     1 (VAR(16),CB),
     1 (VAR(17),V1),
     1 (VAR(18),DB),
     1 (VAR(19),V2),
     1 (VAR(20),EB),
     1 (VAR(21),V3)
      EQUIVALENCE (VAR(22),FB),
     1 (VAR(23),GB),
     1 (VAR(24),ABF),
     1 (VAR(25),XKTPS),
     1 (VAR(26),TBAR),
     1 (VAR(27),STPLAN),
     1 (VAR(28),CL),
     1 (VAR(29),STWET),
     1 (VAR(30),XKLAND),
     1 (VAR(31),WM),
     1 (VAR(32),WS)
      EQUIVALENCE (VAR(33),E1),
     1 (VAR(34),TVAC1),
     1 (VAR(35),XENG1),
     1 (VAR(36),E2),
     1 (VAR(37),TVAC2),
     1 (VAR(38),XENG2),
     1 (VAR(39),XKRCS),
     1 (VAR(40),XLREF),
     1 (VAR(41),AOMS)
      EQUIVALENCE (VAR(42),TVOMS),
     1 (VAR(43),BOMS),
     1 (VAR(44),WPOMS),
     1 (VAR(45),COMS),
     1 (VAR(46),APRIME),
     1 (VAR(47),AC),
     1 (VAR(48),BPRIME),
     1 (VAR(49),CPRIME),
     1 (VAR(50),XKECD),
     1 (VAR(51),AHCD)
      EQUIVALENCE (VAR(52),BHCD),
     1 (VAR(53),ASC),
     1 (VAR(54),BSC),
     1 (VAR(55),WAV),
     1 (VAR(56),VEC),
     1 (VAR(57),DAY),
     1 (VAR(58),AFERP),
     1 (VAR(59),BFERP),
     1 (VAR(60),XKMAF),
     1 (VAR(61),EFER)
      EQUIVALENCE (VAR(62),FPER),
     1 (VAR(63),LPA-FP),
     1 (VAR(64),LPA-LD),
     1 (VAR(65),XKPLF),
     1 (VAR(66),XKOMSR),
     1 (VAR(67),XKRCSR),
     1 (VAR(68),XKRCSE),
     1 (VAR(69),XKOMSC),
     1 (VAR(70),XKRCSC),
     1 (VAR(71),WPAYDE)
      EQUIVALENCE (VAR(72),ARR),
     1 (VAR(73),BRR),
     1 (VAR(74),XKIFL),
     1 (VAR(75), WAPRCP)
      EQUIVALENCE (VAR(76),AICODE)
      EQUIVALENCE (VAR(77),XNEO)
      EQUIVALENCE (WWING,UT(1)),
     1(WTAIL,UT(2)),
     2(WBODY,UT(3)),
     3(WTPS,UT(4)),
     4(WLDR,UT(5)),
     5(WPROP,UT(6)),
     6(WRCS,UT(7)),
     7(WOMS,UT(8)),
     8(WPRIME,UT(9)),
     9(WECD,UT(10))
      EQUIVALENCE (WHCD,UT(11)),
     1(WSC,UT(12)),
     2(WAV2,UT(13)),
     3(WENV,UT(14)),
     4(WPERP,UT(15)),
```

APPENDIX D

EXAMPLE STRUCTURE (EN 155)

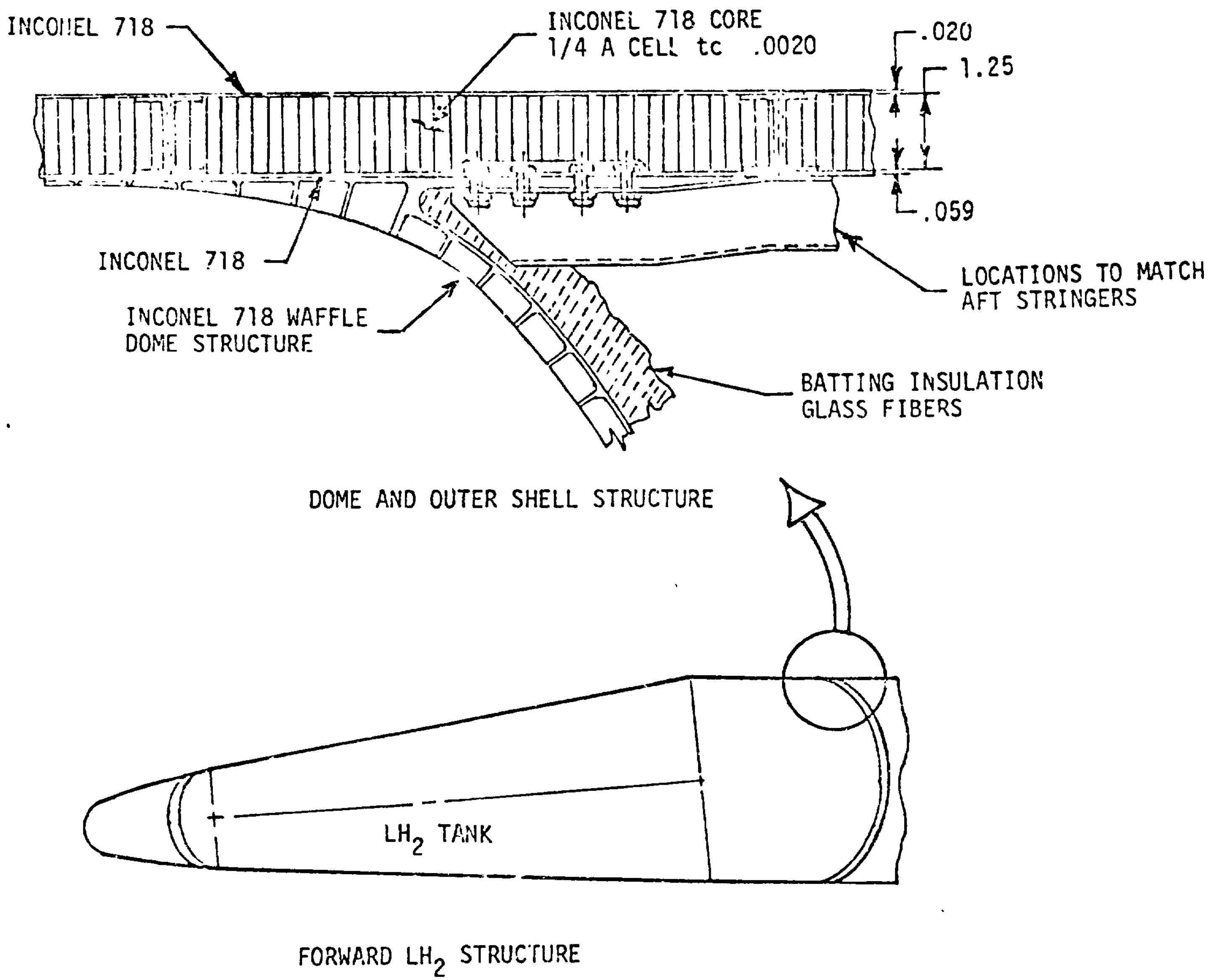

Figure D-1.- Forebody tank on in-house design is designed for Inconel 718 honeycomb. Top portions of tank, where heating rates are lower, were exposed. A thermal protection fairing is used on bottom surfaces.

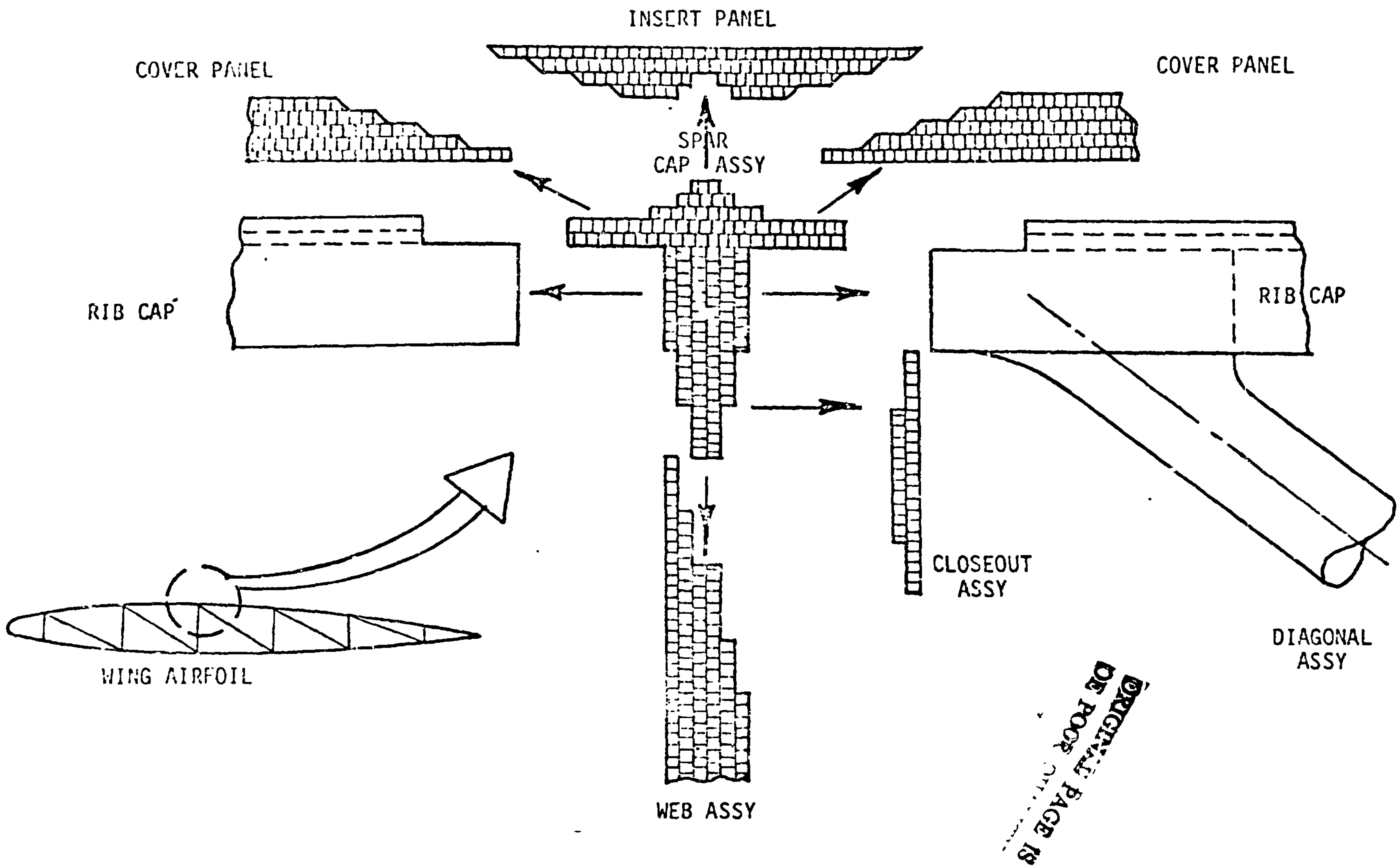

Figure D-2.- Wing is designed for diffusion bonded Inconel 718. Outer layers of honeycomb are nonstructural and used for thermal protection. Lightweight thin-walled honeycomb stabilized tubes are used for internal bracing.

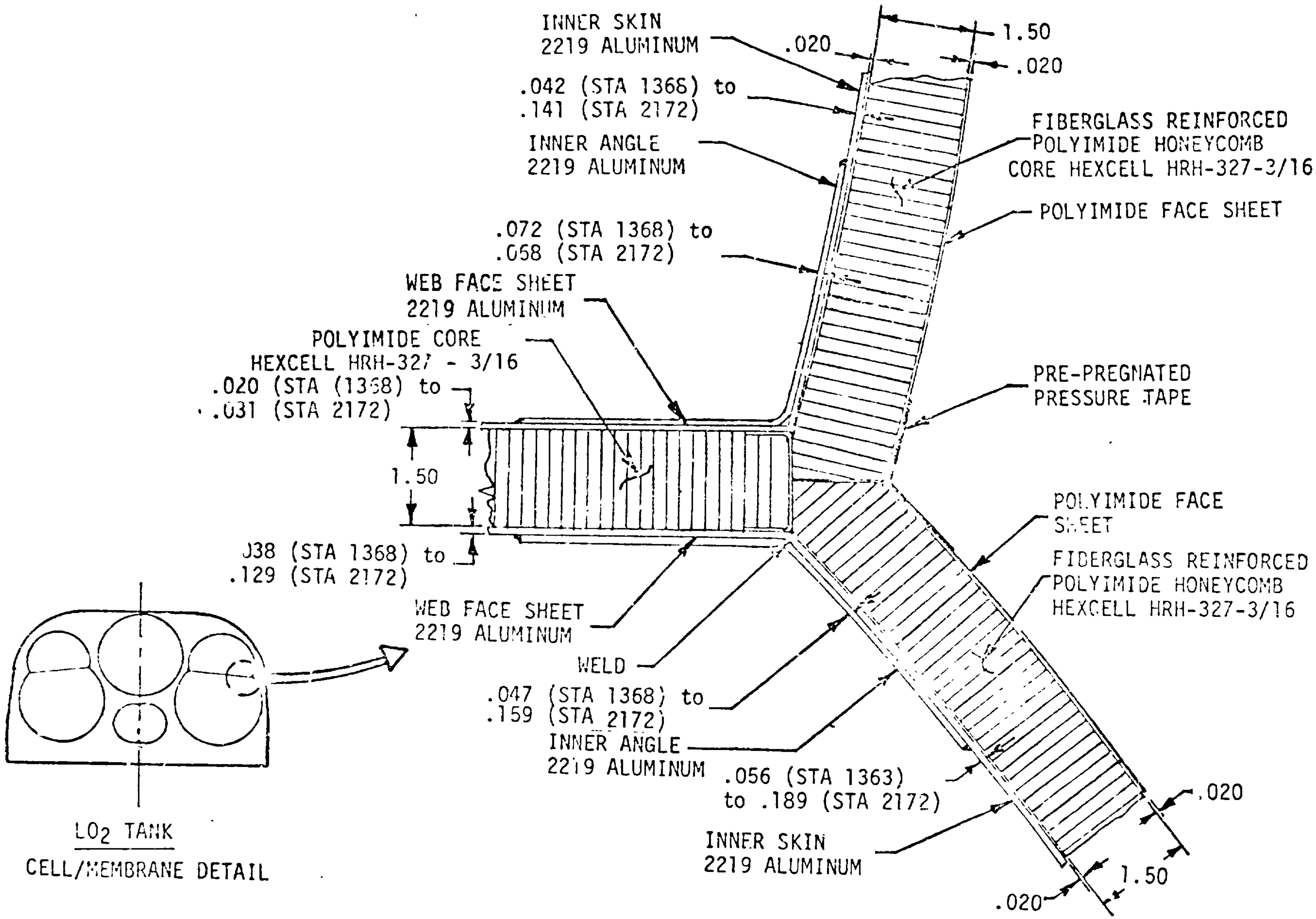

Figure D-3.- For the LO_2 tank, honeycomb wall duals as insulation and for tank wall stabilization

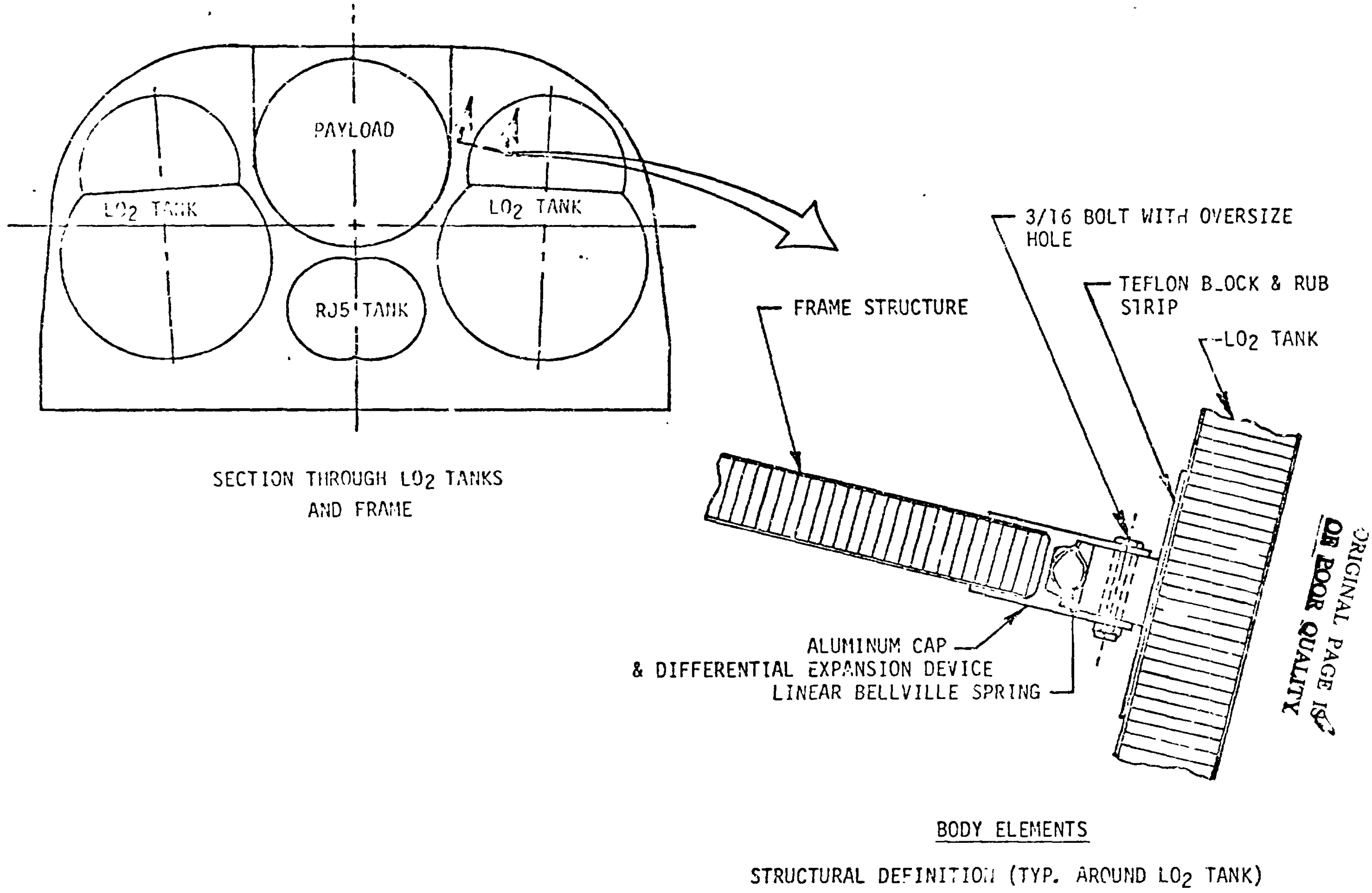

Figure D-4.- Provisions were made in the nonintegral LOX tank installations for axial and radial expansions and contractions

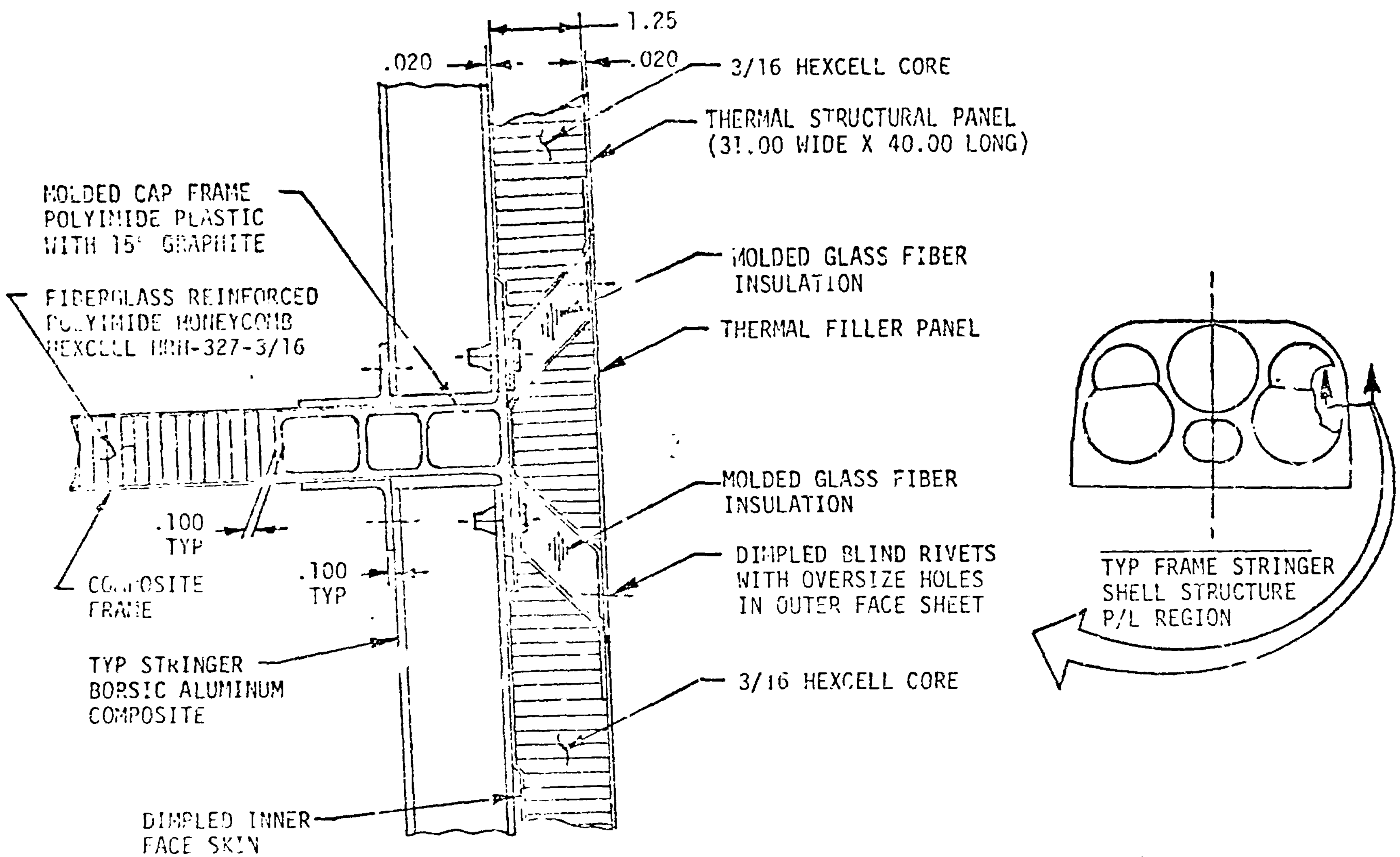

Figure D-5.- Honeycomb panels were selected to carry body shear loads and provide thermal protection in this region of moderate heat loads above the wing surface.

APPENDIX E

EXAMPLE VEHICLE CROSSECTIONS

C-2

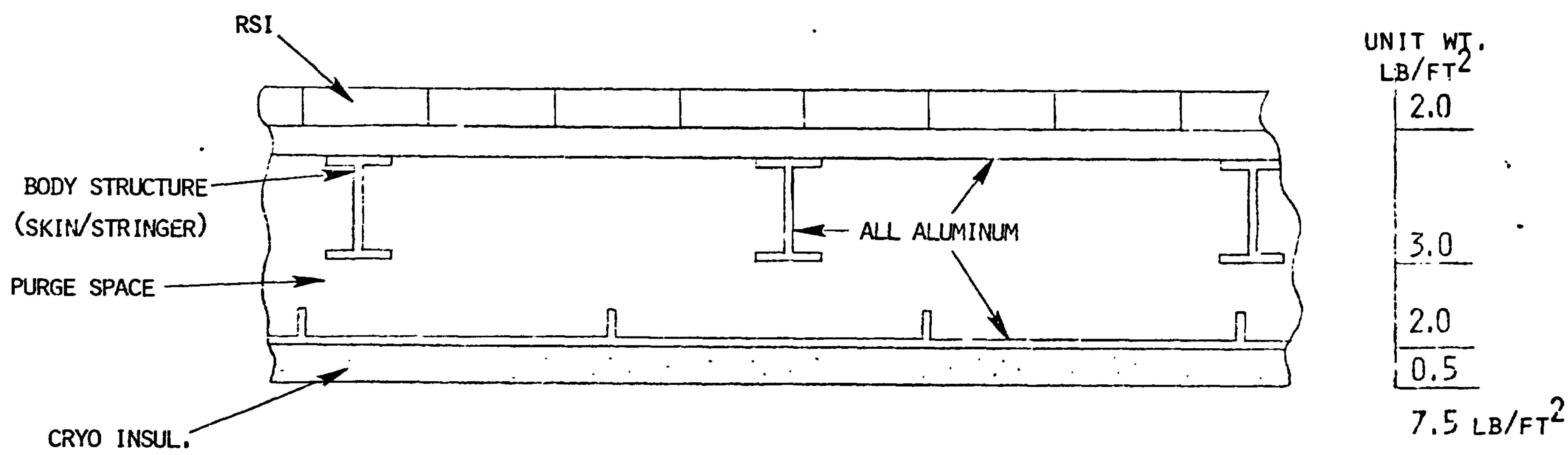

Figure E-1.- A hypothetical body section for hydrogen containment and peak entry tempe1_tures of 1800°F. External and internal insulations, tank, and structure are separated elements.

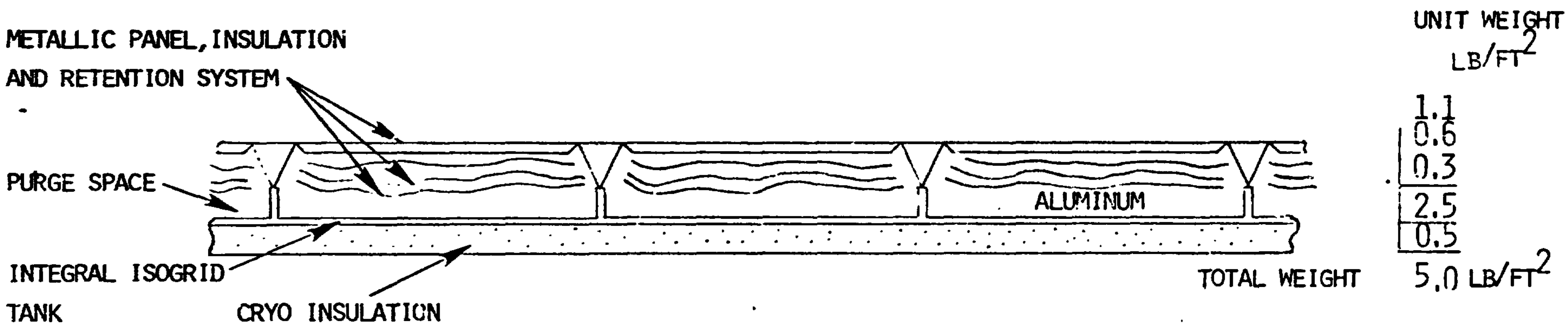

Figure E-2.- In the above figure, fluid containment and body loads are combined into one functional element.

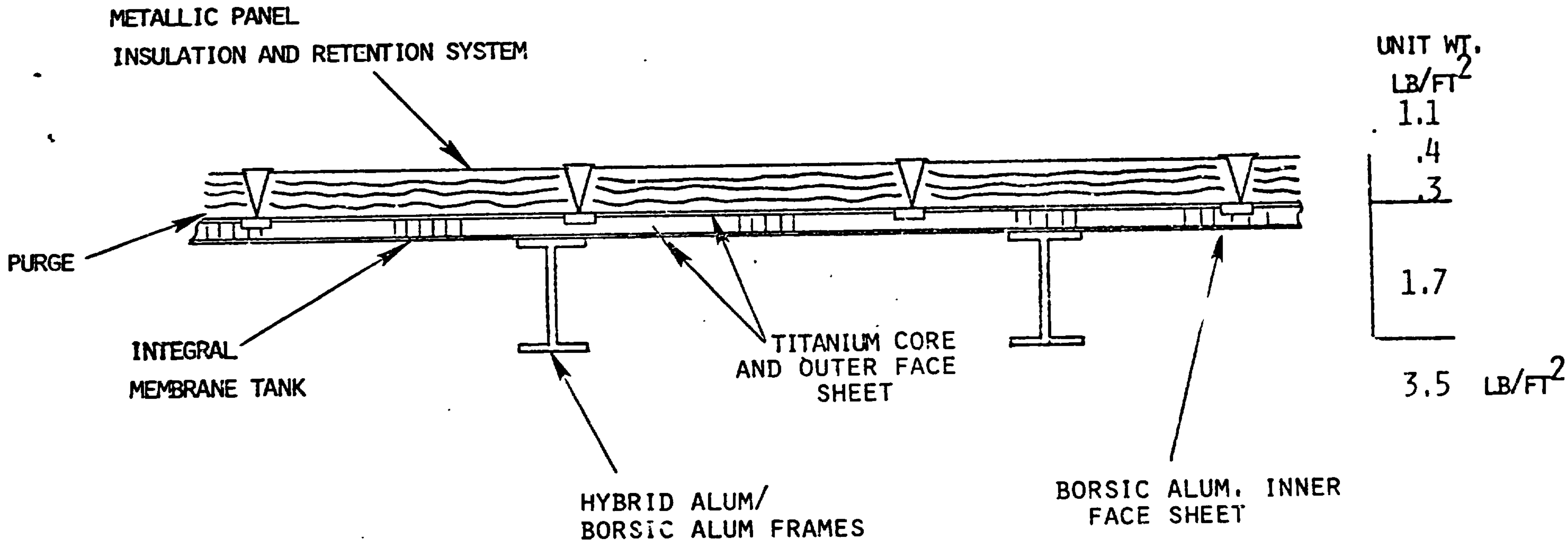

Figure E-3.- In the above system, the cryogenic insulation function has been added to the tank/body element by utilizing honeycomb.

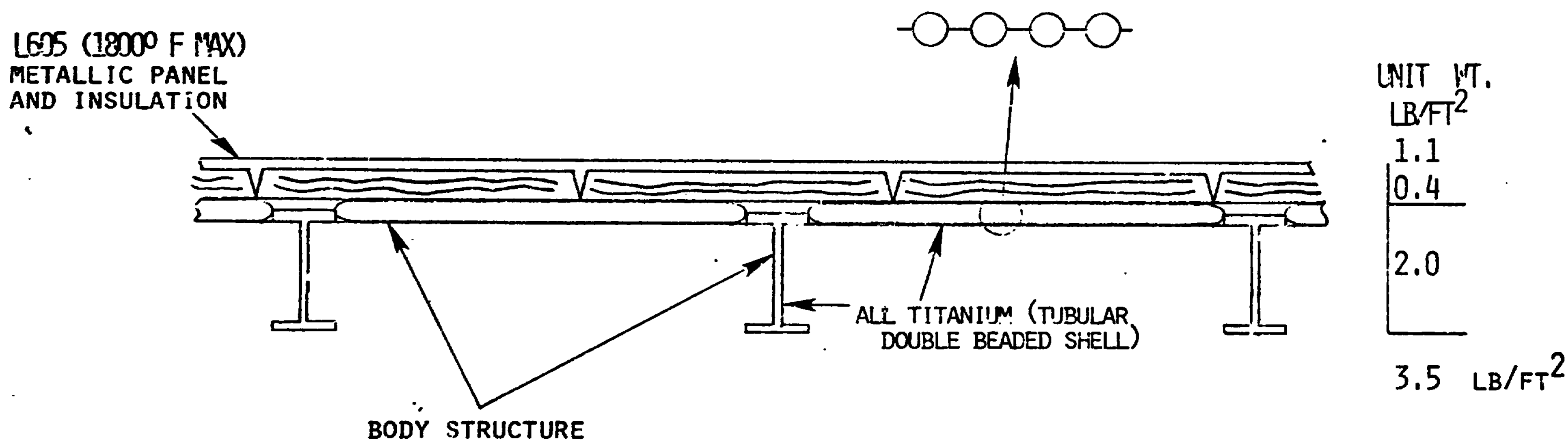

Figure E-4.- Above crossection represents a hypothetical body element in a noncontainment region such as an aft skirt or intertank adapter.

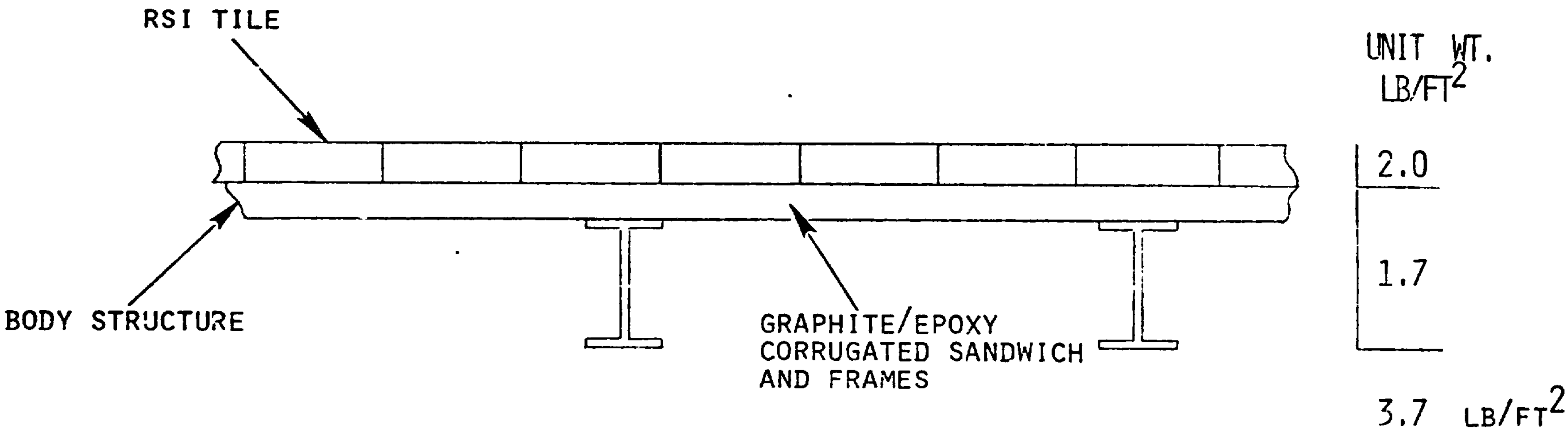

Figure E-5.- The above crossection might represent an area in a similar application to that shown in the previous figure. No strain isolator is utilized between RSI and structure since the two are strain compatible.

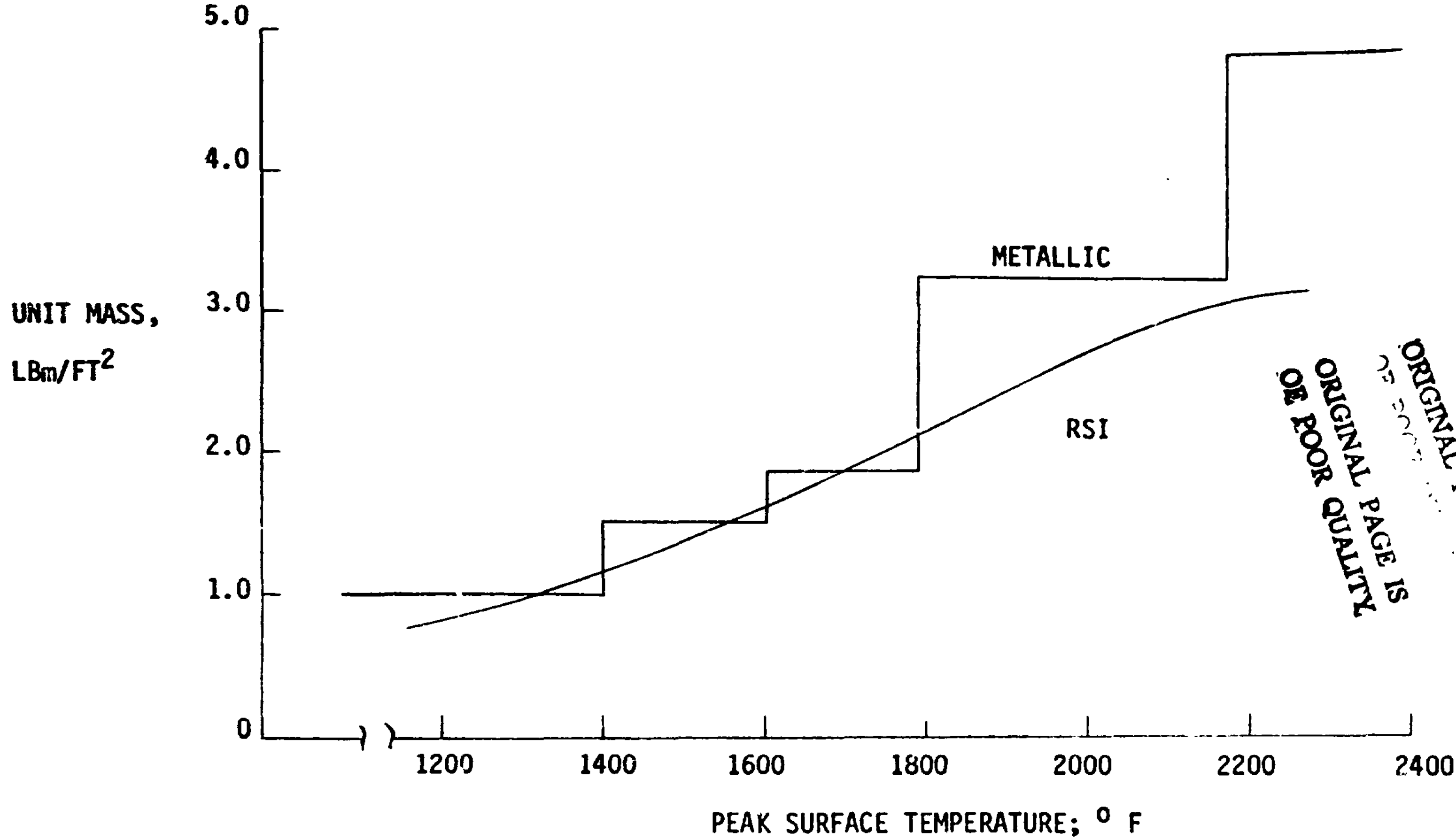

FIGURE E-6.- Thermal protection unit masses versus peak temperatures based on shuttle trajectories.